COPY NO. [redacted]

LMSC-A604141 SP-129-64-17 30 JUNE 1964

LMSC-A604141

PROPULSION SUBSYSTEM

ENGINEERING ANALYSIS REPORT
GEMINI-AGENA TARGET VEHICLE

LIBRARY COPY

JUL 28 1964

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

CLASSIFICATION CHANGE

TO = UNCLASSIFIED

By authority of [redacted] GDS GP-4

[redacted] Date 9/30/75

(NASA-CR-96963) PROPULSION SUBSYSTEM. ENGINEERING ANALYSIS REPORT, GEMINI-AGENA TARGET VEHICLE (Lockheed Missiles and Space Co.) 260 p

N79-76535

Unclas
00/20 11091

FAC (NASA CR OR TMX OR AD NUMBER) (CATEGORY)

U. S. Government Agencies and Contractors Only

Lockheed

MISSILES & SPACE COMPANY

A DIVISION OF LOCKHEED AIRCRAFT CORPORATION

SUNNYVALE, CALIFORNIA

[redacted]

Report COPY NO. 8

In reply refer to:
LMSC/A753133
Orgn. 67-10
3F33

LOCKHEED
MISSILES
& SPACE
COMPANY

MGR
DM
SRA
TA
SA
AA
GP
GS
GF
OS
F

JUN 11 10 57 AM '65

RECEIVED
GEMINI FILES

8 June 1965

Subject: Upgrade Action

To: NASA Manned Spacecraft Center
Attn: C. W. Matthews
Gemini Program Office
Houston, Texas

Reference: (a) Confidential LMSC report, A604141, "Gemini Propulsion System Engineering Analysis Report" dated 30 June 1964

1. Twenty copies of the referenced report were transmitted to your office on 16 July 1964 under LMSC transmittal letter A604093.

2. Subsequent to the above dispatch, it was determined by the Lockheed Missiles & Space Company, Space Systems Division Classification Office that page 6-15 of the report should be upgraded to Confidential.

3. In accordance with the above, it is requested that your office take the necessary steps to complete this upgrade action.

LOCKHEED MISSILES & SPACE COMPANY

J. R. Lulis
Document Control Station-S

JRL:fo

accomplished 1 July 65
ga

~~CONFIDENTIAL~~

C68-4279

LMSC-A604141

Downgraded at 3 year intervals ... Classified ... years

PROPULSION SUBSYSTEM

ENGINEERING ANALYSIS REPORT
GEMINI-AGENA TARGET VEHICLE

(UNCLASSIFIED TITLE)

DOWNGRADED ... DECLASSIFIED AFTER 12 YEARS ... DOD DIR 5200.10

Prepared Under Authority of Paragraph 2.2.2

Contract AF-04(695)-129

This document contains information affecting the national defense of the United States within the meaning of the Espionage Laws, Title 18, U. S. C., Sec. 793 and 794. Its transmission or the revelation of its contents in any manner to an unauthorized person is prohibited by law.

APPROVED:

M. J. Mercer

for J. J. KENNELLY, MANAGER
MSVP ENGINEERING

APPROVED:

J. L. Shoenhair

J. L. SHOENHAIR, MANAGER
MEDIUM SPACE VEHICLE PROGRAMS

Lockheed

MISSILES & SPACE COMPANY

A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION

SUNNYVALE, CALIFORNIA

~~CONFIDENTIAL~~

FOREWORD

This Propulsion Subsystem Engineering Analysis Report is produced by Lockheed Missiles and Space Company for the National Aeronautics and Space Agency and Air Force and is prepared under authority of Paragraph 2.2.2, Contract AF-04(695)-129. The document describes and presents an analysis of the Propulsion Subsystem for the Gemini-Agena Target Vehicle.

CONTENTS

Section		Page
	FOREWORD	iii
	ILLUSTRATIONS	vii
	TABLES	xi
	ABBREVIATIONS	xiii
1	INTRODUCTION	1-1
	1.1 Scope	1-1
	1.2 Vehicle Configuration and Definition of Subsystem	1-2
	1.3 Mission Requirements	1-4
	1.4 Performance and Environment Specifications	1-6
2	DESCRIPTION OF GEMINI-AGENA TARGET VEHICLE SUBSYSTEM B	2-1
	2.1 General Description of Subsystem	2-1
	2.2 Main Engine	2-5
	2.3 Propellant Feed and Load System	2-52
	2.4 Gas Pressurization System Assembly	2-64
3	PROPELLANTS AND PROPELLANT LOADINGS	3-1
	3.1 Rocket Engine Propellants	3-1
	3.2 Other Mediums	3-4
	3.3 Propellant Loading	3-6
4	ENGINE OPERATION	4-1
	4.1 Ullage Orientation Control	4-1
	4.2 Engine Start Sequence (Flight)	4-9
	4.3 Engine Shutdown (Normal Flight)	4-12
	4.4 Rocket Engine Ground Operation	4-14

CONTENTS (Cont)

Section		Page
5	INSTRUMENTATION	5-1
	5.1 Introduction	5-1
	5.2 Preliminary Instrumentation Requirements	5-2
	5.3 Flight Instrumentation – Telemetry	5-9
	5.4 Agena Status Display Panel	5-24
	5.5 Pressure Switches	5-36
6	PRIMARY PROPULSION SYSTEM PERFORMANCE AND DESIGN CHARACTERISTICS	6-1
	6.1 Introduction	6-1
	6.2 Engine Requirements	6-1
	6.3 Orifice-Fed Pressurization System	6-3
	6.4 Subsystem Performance in Vacuum Flight	6-5
	6.5 Engine Vacuum Performance	6-9
	6.6 Nominal Performance Data	6-13
	6.7 Exhaust Duct Thrust	6-14
	6.8 Rocket Engine Performance Prediction (Preflight)	6-14
	6.9 Rocket Engine Performance Analysis (Postflight)	6-23
	6.10 Estimated Engine Altitude Performance	6-37
7	PYROTECHNICS	7-1
	7.1 Introduction	7-1
	7.2 Devices Used	7-1
8	AEROSPACE GROUND EQUIPMENT	
	8.1 Introduction	8-1
	8.2 Primary Propulsion System Checkout Console	8-1
	8.3 Start Tank Loading Carts	8-12
	8.4 Auxiliary Flushing Unit (Start Tank Flushing)	8-16
	8.5 Vacuum Start Tank Bake-Out Oven	8-17
	8.6 Main System Flushing	8-17
BIBLIOGRAPHY		B-1

ILLUSTRATIONS

Figure		Page
1-1	Gemini-Agena Target Vehicle Configuration	1-3
1-2	Ascent Sequence of Events for Gemini-Agena Target Vehicle	1-5
2-1	Gemini-Agena Target Vehicle Inboard Profile	2-2
2-2	Primary Propulsion System Fluid Flow Schematic	2-3
2-3a	Rocket Engine – Left Side View	2-6
2-3b	Rocket Engine – Right Side View	2-6
2-4	Rocket Engine – View Looking Aft	2-7
2-5	Rocket Engine Electrical Schematic Diagram	2-8
2-6	Electrical Gate Overspeed Circuit Block Diagram	2-9
2-7	Mounting of Motional Pickup Transducer	2-11
2-8	Fuel Start Tank	2-15
2-9	Fuel Start Tank Manual Fuel Bleed Valve	2-17
2-10	High-Pressure Nitrogen Charging Valve (Start Tank)	2-17
2-11	Fuel Start Tank Fill and Drain Valve	2-19
2-12	Oxidizer Start Tank	2-20
2-13	Oxidizer Start Tank Fill and Drain Check Valve	2-21
2-14	Fuel and Oxidizer Venturi and Filter Assembly	2-23
2-15	Oxidizer Dual Check Valve	2-24
2-16	Fuel Dual Check Valve	2-25
2-17	Gas Generator Fuel Solenoid Valve	2-27
2-18	Gas Generator Oxidizer Solenoid Valve	2-29
2-19	Turbopump Assembly	2-33
2-20	Cross Section of Fuel Pump	2-34
2-21	Cross Section of Oxidizer Pump	2-34
2-22	Vapor Pressure Versus Temperature (UDMH and IRFNA)	2-36

ILLUSTRATIONS (Cont)

Figure		Page
2-23	Pilot-Operated Solenoid Valve	2-38
2-24	Fuel Valve	2-39
2-25	Pilot-Operated Solenoid Valve and Fuel Valve	2-41
2-26	Oxidizer Valve	2-42
2-27	Thrust Chamber Assembly	2-44
2-28	Thrust Chamber Nozzle Extension	2-46
2-29	Cross Section of Engine Injector	2-48
2-30	View Showing Engine Injector	2-49
2-31	Engine Gimbal Attachment and Movement	2-51
2-32	Propellant Tank Assembly	2-53
2-33	Fuel Sump (Propellant Containment System)	2-55
2-34	Oxidizer Sump (Propellant Containment System)	2-56
2-35	Propellant Isolation Valve	2-58
2-36	Airborne Portion of Propellant Tank Fill and Drain Disconnect	2-61
2-37	Propellant Pressurization System	2-65
2-38	Airborne Portion of Propellant Tank Vent Disconnect	2-67
2-39	Helium Sphere Mounting Method	2-68
2-40	Pyrotechnically Operated Helium-Control Valve — Normal Condition	2-70
2-41	Pyrotechnically Operated Helium-Control Valve — First Actuation	2-70
2-42	Pyrotechnically Operated Helium-Control Valve — Second Actuation	2-71
3-1	Specific Gravity Versus Temperature (UDMH and IRFNA)	3-2
3-2	Fuel and Oxidizer Start Tank Nitrogen Fill Ports	3-20
3-3	Engine Start Tank System	3-21
3-4a	Fuel to be Loaded at $40^{\circ} \pm 5^{\circ}F$	3-24
3-4b	Fuel to be Loaded at $50^{\circ} \pm 5^{\circ}F$	3-24
3-4c	Fuel to be Loaded at $60^{\circ} \pm 5^{\circ}F$	3-25
3-4d	Fuel to be Loaded at $70^{\circ} \pm 5^{\circ}F$	3-25
3-4e	Fuel to be Loaded at $80^{\circ} \pm 5^{\circ}F$	3-26

ILLUSTRATIONS (Cont)

Figure		Page
3-4f	Fuel to be Loaded at 90° ±5°F	3-26
3-4g	Fuel to be Loaded at 100° ±5°F	3-27
3-4h	Fuel to be Loaded at 110° ±5°F	3-27
3-5a	Oxidizer to be Loaded at 40° ±5°F	3-28
3-5b	Oxidizer to be Loaded at 50° ±5°F	3-28
3-5c	Oxidizer to be Loaded at 60° ±5°F	3-29
3-5d	Oxidizer to be Loaded at 70° ±5°F	3-29
3-5e	Oxidizer to be Loaded at 80° ±5°F	3-30
3-5f	Oxidizer to be Loaded at 90° ±5°F	3-30
3-5g	Oxidizer to be Loaded at 100° ±5°F	3-31
3-5h	Oxidizer to be Loaded at 110° ±5°F	3-31
4-1	Total Propellant Orientation Period Prior to Main Engine Ignition Versus Dimensionless Vehicle Acceleration	4-3
4-2	Secondary Propulsion Module – Aft View	4-5
4-3	Secondary Propulsion System Fluid Flow Schematic	4-6
5-1	Primary Propulsion System Prelaunch Helium Loading	5-4
5-2	Secondary Propulsion System Prelaunch Nitrogen Loading	5-8
5-3	Engine Switch Group Nominal Voltage Levels	5-14
5-4	Agena Status Display Panel	5-25
5-5	Primary Propulsion System and Secondary Propulsion System Status Display Panel	5-28
6-1	Fuel Pump Minimum Inlet Pressure Versus Propellant Temperature	6-2
6-2	Oxidizer Pump Minimum Inlet Pressure Versus Propellant Temperature	6-2
6-3	Helium Sphere and Propellant Tank Pressure-Time Histories – Squib-Operated Helium-Control Valve	6-4
6-4	Liquid Head Versus Volume for Propellant Tanks	6-8
6-5	Variation of Rated Thrust With Altitude	6-10
6-6	Estimated Rocket Engine Thrust Limits Over Operating Requirements	6-11

ILLUSTRATIONS (Cont)

Figure		Page
6-7	Variation of Rocket Engine Minimum Overall Specific Impulse With Altitude	6-12
6-8	Oxidizer Pump Flow Versus Total Suction Pressure	6-16
6-9	Fuel Pump Flow Versus Total Suction Pressure	6-17
6-10	Oxidizer Pump Static Pressure Rise Versus Flow Rate at Constant Speeds	6-18
6-11	Fuel Pump Static Pressure Rise Versus Flow Rate at Constant Speeds	6-19
6-12	Oxidizer Pump Flow Cavitation Characteristic	6-26
6-13	Fuel Pump Flow Cavitation Characteristic	6-27
7-1	Typical Shaped Charge – Cone End	7-8
7-2	Shaped Charge	7-9
7-3	Destructor Initiator	7-10
7-4	Shaped Charge and Initiator Assembly	7-11
7-5	Retrorocket – Cutaway and End View	7-15
7-6	Separation Pin Pusher – Horizon Sensor Fairing-Vehicle	7-19
7-7	Horizon Sensor Torque Tube Actuating Pin Pusher	7-20
7-8	Explosive Shroud Bolt	7-22
8-1	Primary Propulsion System Checkout Console	8-2
8-2	Oxidizer Start Tank Loading Cart	8-3
8-3	Fuel Start Tank Loading Cart	8-4
8-4	Start Tank Auxiliary Flushing Unit – (Front View)	8-5
8-5	Start Tank Auxiliary Flushing Unit – (Back View)	8-6
8-6	Vacuum Bake-Out Oven (Start Tank)	8-7
8-7	Pneumatic Portion of Primary Propulsion System Checkout Console	8-9
8-8	Electric Portion of Primary Propulsion System Checkout Console	8-10
8-9	Oxidizer Start Tank Servicing Unit	8-13
8-10	Fuel Start Tank Servicing Unit	8-15

TABLES

Table		Page
2-1	Design Characteristics of Gas Generator Fuel Solenoid Valve	2-28
2-2	Design Characteristics of Gas Generator Oxidizer Solenoid Valve	2-30
2-3	Propellant Isolation Valve Operating Characteristics	2-60
2-4	Helium Sphere Specifications	2-69
3-1	Cleanliness Requirements of Pneumatic Fluid Media	3-4
3-2	Transient- and Non-Impulse Propellants Initial Burns	3-7
3-3	Transient- and Non-Impulse Propellants Final Burn	3-8
3-4	Residuals (Trapped Non-Impulse Propellants)	3-9
3-5	Nominal Tank Volumes	3-9
3-6	Quantity of UDMH to be Loaded Into Fuel-Start Tank	3-19
3-7	Quantity of IRFNA to be Loaded Into Oxidizer-Start Tank	3-23
5-1	Primary Propulsion System Prelaunch Instrumentation Requirements (Landline)	5-2
5-2	Secondary Propulsion System Prelaunch Instrumentation Requirements (Landline)	5-7
5-3	Primary Propulsion System Flight Instrumentation	5-9
5-4	Secondary Propulsion System Flight Instrumentation	5-18
5-5	Pressure Switches Design Characteristics	5-37
6-1	Rocket Engine Performance at Altitude	6-9
6-2	Estimated Engine Altitude Performance and Overall Characteristics	6-37

TABLES (Cont)

Table		Page
7-1	Gemini Pyrotechnics	7-2
7-2	Cartridge Electrical Characteristics	7-3
7-3	Separation Detonator Electrical Characteristics	7-5
7-4	Self-Destruct System Detonator Electrical Characteristics	7-13
7-5	Performance Characteristics of 0.9 KS-500 Retrorockets	7-14
7-6	Nominal Weights and Dimensions of Retrorockets	7-16
7-7	Booster Retrorocket Electrical Characteristics	7-17
7-8	Separation Bolt Detonating Cartridge Electrical Characteristics	7-21

ABBREVIATIONS

AGE	Aerospace Ground Equipment
AMR	Atlantic Missile Range
ASP	Agena Status Panel (Display)
ATV	Agena Target Vehicle
BAC	Bell Aerosystems Company
cch	Cubic centimeters per hour
cps	Cycles per second
EMI	Electromagnetic Interferance
°F	Degree Fahrenheit
f_{out}	Motional pickup transducer pulse rate output
gpm	Gallons per minute
GGFSV	Gas generator fuel solenoid valve
GGMPS	Gas generator manifold pressure switch
GGOSV	Gas generator oxidizer solenoid valve
GG	Gas generator
IRFNA	Inhibited red fuming nitric acid
lbf	Pounds force
lbm	Pounds mass
LSB	Least significant bit
MON	Mixed oxides of nitrogen
nm	Nautical mile(s)
N_2	Nitrogen gas
N_t	Turbine speed
OFPS	Oxidizer feed pressure switch
POSV	Pilot-operated solenoid valve
POHCV	Pyrotechnically operated helium-control valve
P	Pressure
P_c	Thrust chamber pressure

ABBREVIATIONS (Continued)

psia	Pounds per square inch absolute
psid	Pounds per square inch difference
psig	Pounds per square inch gage
PIV	Propellant isolation valve
POSV	Pilot-operated solenoid valve
ppm	Parts per million
PPS	Primary Propulsion System
PTVA	Propellant Test Vehicle Assembly
QD	Quick Disconnect
rpm	Revolutions per minute
RMD	Reaction Motors Division (Thiokol Chemical Corporation)
S/C	Spacecraft
SPS	Secondary Propulsion System
TCPS	Thrust chamber pressure switch
TLM	Telemetry
UDMH	Unsymmetrical dimethylhydrazine
V_{pp}	Volts, peak-to-peak
vdc	Volts, direct current
WCG	Whittaker Controls and Guidance (Division of Telecomputing Corporation

Section 1
INTRODUCTION

1.1 SCOPE

This report presents operational modes and performance capability analysis of the NASA Agena Target Vehicle (ATV) utilized in the Gemini program. The propulsion subsystem B is defined, mission requirements are set forth, and performance and environmental specifications are listed in this introductory section.

- Section 2 concerns the overall Primary Propulsion System (PPS), presents a breakdown of the subsystems, and describes in detail the purpose and operation of each component.

- Section 3 discusses the propellants used and the method of loading these propellants. Also, the method for calculating the required quantity of propellants to be loaded into the main and start tanks is presented.

- Section 4 describes general engine operation. Ullage orientation, including the need for and means of implementation and other special requirements, is discussed.

- Section 5 considers instrumentation requirements affecting the operation of the PPS. Three categories are covered as follows:

    1. Prelaunch instrumentation which gives "Launch" or "Hold" information
    2. Flight instrumentation which, through telemetry, provides information necessary for evaluating total system and flight performance

3. Instrumentation on the Agena Target Vehicle Status Display Panel, which imparts information concerning the status of the ATV for the final docking phase and for PPS and Secondary Propulsion System (SPS) operation by the astronaut subsequent to docking.

• Section 6 discusses the propulsion subsystem performance capability and the parameters governing engine performance and power level in vacuum flight. The equations used for computing engine-power level are general in that they include all significant parameters known at this time. As continued experience is gained, periodic adjustment of these equations may be expected.

• Section 7 describes and details the pyrotechnic devices together with pyrotechnic checkout requirements as used in the Agena Target Vehicle.

• Section 8 of this report briefly lists and describes the Aerospace Ground Equipment (AGE) peculiar to the checkout and servicing of the Model 8247 rocket engine.

1.2 VEHICLE CONFIGURATION AND DEFINITION OF SUBSYSTEM

The Gemini-Agena Target Vehicle design is an adaptation of the basic Agena-D vehicle using the alternate Model 8247 rocket engine and additional program-peculiar equipment required for the Gemini mission. This ATV is essentially divided into (1) the program-peculiar forward auxiliary section, (2) the Agena-D forward and midbody sections, and (3) the program-peculiar aft section. The intent of this report is to discuss only the systems applicable to subsystem B and its operations.

Vehicle Configuration

The auxiliary forward section (Fig. 1-1) consists of the auxiliary equipment rack, the McDonnell Aircraft Company-furnished docking-adapter module, and the clamshell nose shroud. The Agena-D forward section

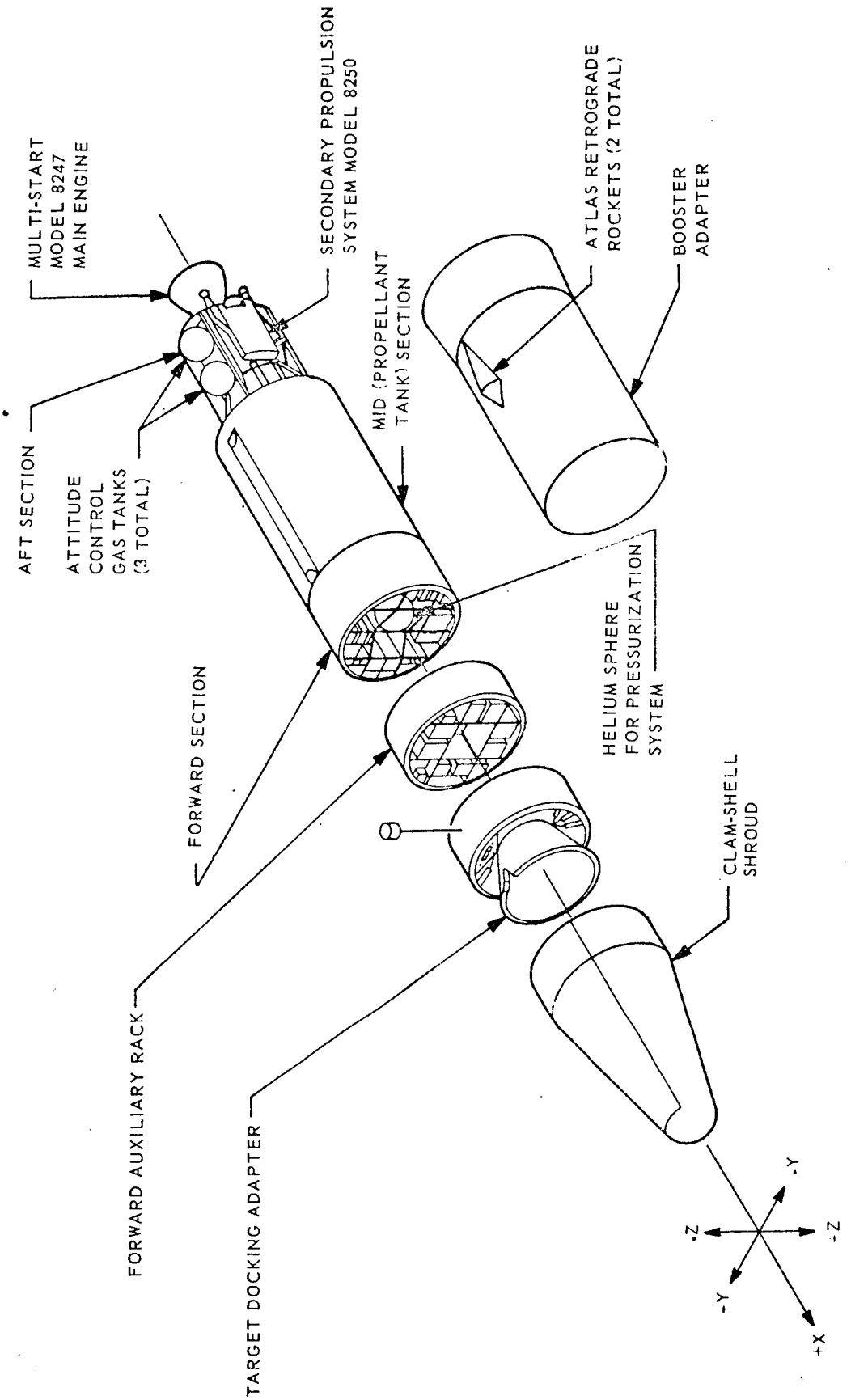


Fig. 1-1 Gemini-Agena Target Vehicle Configuration

houses the main equipment bay, and the midbody contains the main fuel and oxidizer tanks which supply propellants through a feed and load system for the main engine. The Model 8247 multi-start main engine and the smaller Model 8250 maneuvering and ullage orientation engines are located in the aft section.

Orbital length of the ATV is approximately 26 feet. Vehicle weight-on-orbit is approximately 7200 lb. This weight includes propellants still remaining in the main tanks and available for Model 8247 engine operation after the Agena achieves orbit. The Gemini-ATV propulsion system consists of the following:

- Model 8247 rocket engine, also known as XLR-81-BA-13, and its controls, mount, gimbals, and titanium nozzle extension
- Pyrotechnically operated helium-control valve (POHCV) and associated pressurization plumbing
- Fuel and oxidizer feed and load system, including propellant tanks, vents, and fill quick disconnects
- Propellant isolation valves (PIV's)
- All associated pyro devices and solid-propellant rockets.

1.3 MISSION REQUIREMENTS

The propulsion system is designed to inject the ATV into an orbital path (Fig. 1-2) which has an altitude varying between 161 nautical miles (nm) and 87 nm when measured at a latitude of 28.34 deg. A minimum of five main-engine burns are available to complete the mission requirements; one burn is required for injection into orbit and four subsequent burns are available to perform orbital plane and phase changes, as required and selected from ground stations or from the spacecraft. All launches are presently scheduled to take place from AMR Pad 14.

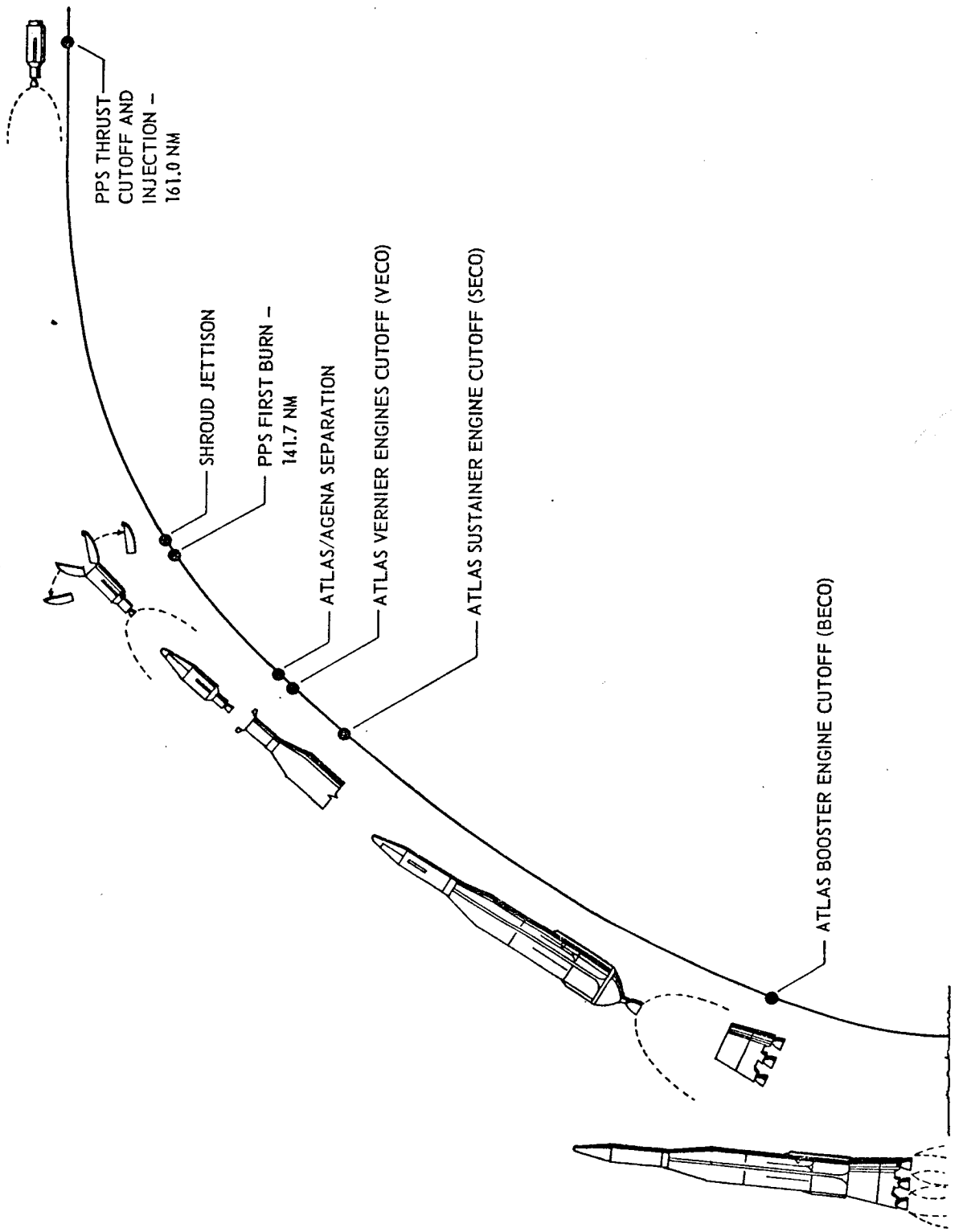


Fig. 1-2 Ascent Sequence of Events for Gemini-Agena Target Vehicle

Once the ATV is in orbit, the PPS provides the thrust necessary for the following maneuvers:

1. Adjusting the orbits so that they are coplanar and making the radius of one apside of the Agena orbit equal to the rendezvous radius
2. Circularizing the Agena at the rendezvous radius and proceeding to an inertial location which will be a point lying on the line of the apsides of the catch-up orbit
3. At the initial location of the injection point, transfering the Agena to a catch-up orbit for placing the vehicle at the rendezvous point when specified.

1.4 PERFORMANCE AND ENVIRONMENT SPECIFICATIONS

The latest performance and environment specifications applicable to this report are as follows:

LMSC or LAC:

1414463	Model Specification, Engine, Rocket, Liquid Propellant USAF Model XLR-81-BA-13 (U)
1067287	Propellant Umbilical Connectors Remote-Operated Quick Disconnect
1412861	Pneumatic Pressure Vessels
1414804	Pyro Helium Control Valve
1415273	Valve Propellant Shutoff, Motor-Operated
1510458	Propellant Vent Umbilical Quick-Disconnect Coupling
1418B	Lines and Fitting, Gas and Liquid, Cleaning of
1414805	Propulsion System Test Requirements for Manufacturing and Final Acceptance Tests
1416537	Requirement Specification Agena Target Vehicle Propulsion System Test Requirement

Federal:

BB-N-411A	Nitrogen, Liquid and Gaseous
MIL-P-25604C	Military Specification Propellant, unsymetrical-dimethyl-hydrazine (UDMH)
MIL-P-7254E	Military Specification Propellant, Nitric Acid.

Section 2
DESCRIPTION OF GEMINI-AGENA TARGET VEHICLE SUBSYSTEM B

2.1 GENERAL DESCRIPTION OF SUBSYSTEM

A general arrangement of the NASA Gemini-Agena Target Vehicle (ATV), including the location and physical relationships of the various propulsion subsystem assemblies, is shown in Fig. 2-1. The entire subsystem is organized into the following major assemblies:

- Main Engine – USAF XLR-81-BA-13 (LMSC-1461969) multi-start rocket engine
- Propellant feed and load system
- Gas pressurization system assembly
- Secondary propulsion system
- Pyrotechnics

In this section, the main engine, the propellant feed and load system, and the gas pressurization system will be discussed in detail so that the engine operation (Section 4) may be better understood. Refer to Fig. 2-2 for the functional relationship of the various individual components.

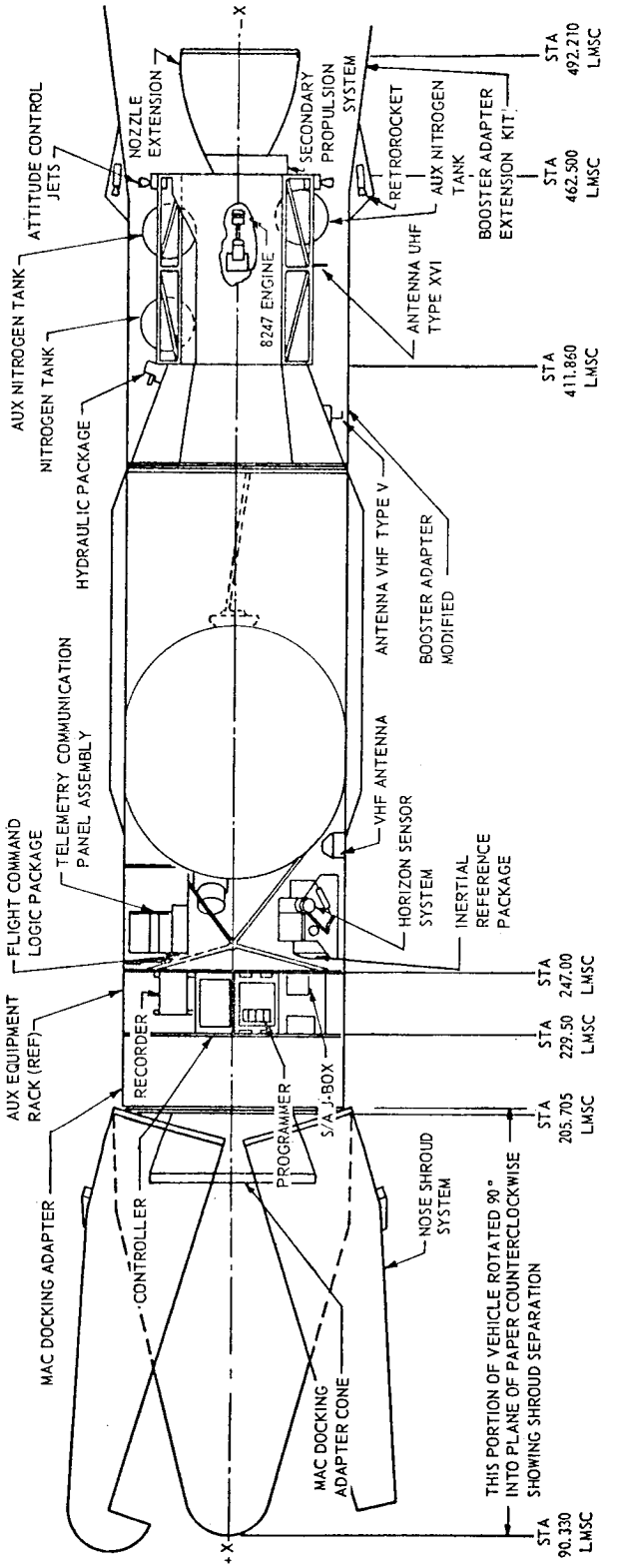
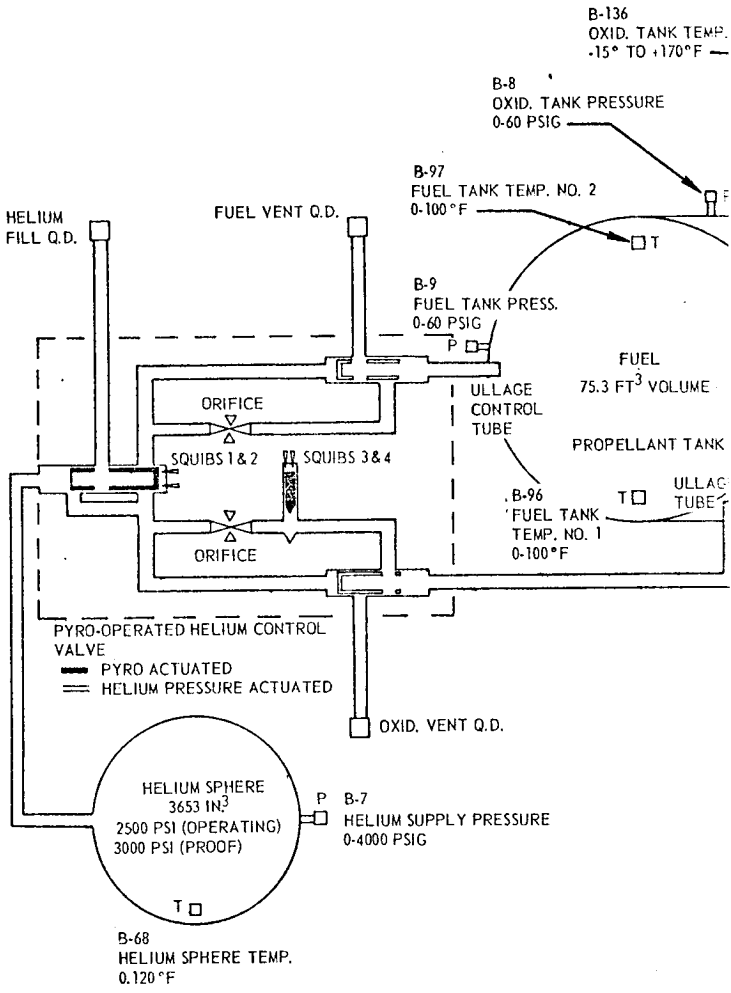


Fig. 2-1 Gemini-Agena Target Vehicle Inboard Profile

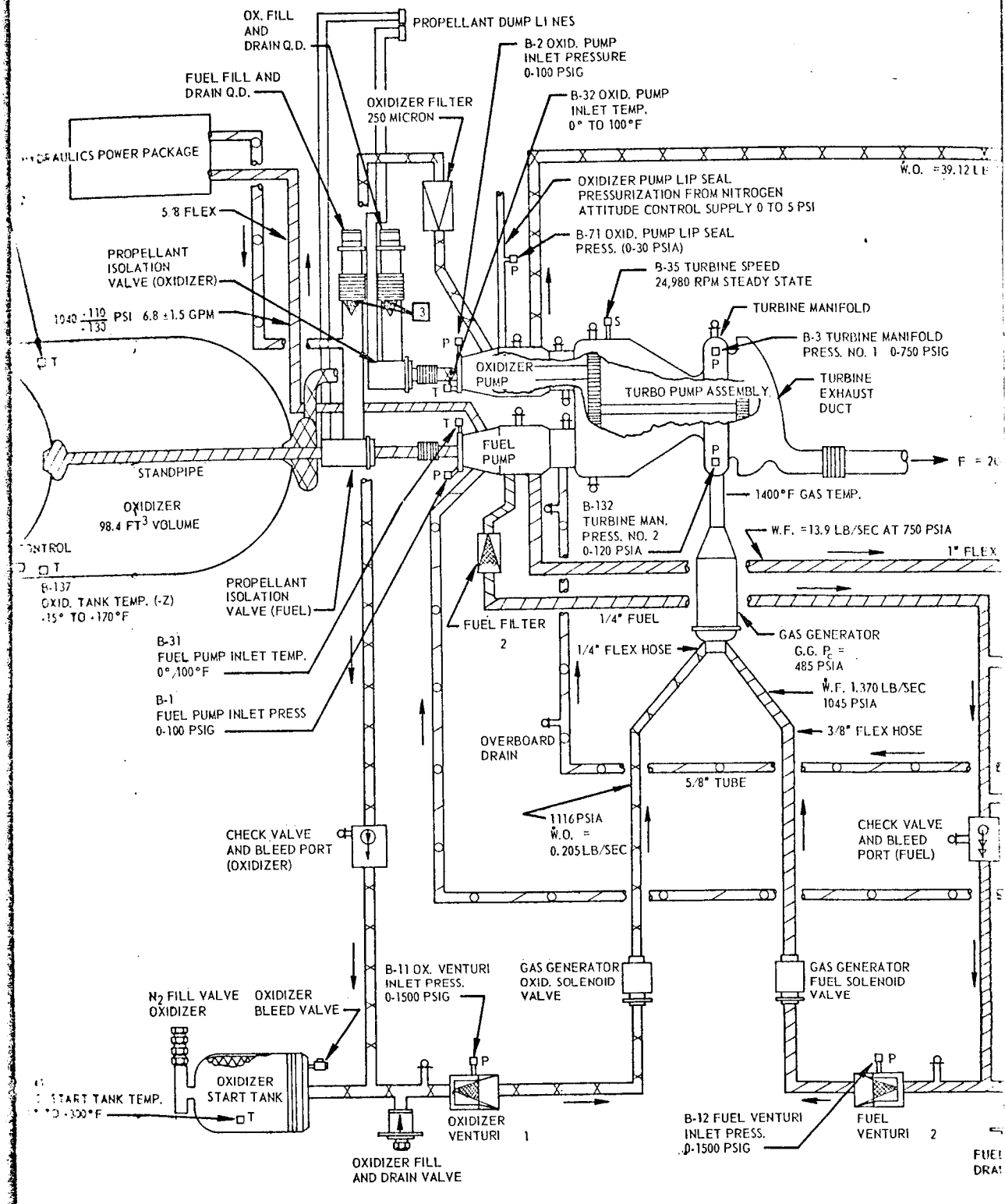


FILTERS

1 WILL NOT PASS PARTICLES GREATER THAN 0.010" DIA.

2 WILL NOT PASS PARTICLES GREATER THAN 0.025" DIA.

3 WILL NOT PASS PARTI CLES GREATER THAN 0.030" DIA.


OX. FILL AND DRAIN Q.D.
PROPELLANT DUMP LINES
B-2 OXID. PUMP INLET PRESSURE 0-100 PSIG
FUEL FILL AND DRAIN Q.D.
OXIDIZER FILTER 250 MICRON
B-32 OXID. PUMP INLET TEMP. 0° TO 100°F
HYDRAULICS POWER PACKAGE
W.O. =39.12 LB
OXIDIZER PUMP LIP SEAL PRESSURIZATION FROM NITROGEN ATTITUDE CONTROL SUPPLY 0 TO 5 PSI
5/8 FLEX
B-71 OXID. PUMP LIP SEAL PRESS. (0-30 PSIA)
PROPELLANT ISOLATION VALVE (OXIDIZER)
B-35 TURBINE SPEED 24,980 RPM STEADY STATE
1040 -110 -130 PSI 6.8 ±1.5 GPM
TURBINE MANIFOLD
B-3 TURBINE MANIFOLD PRESS. NO. 1 0-750 PSIG
OXIDIZER PUMP
TURBO PUMP ASSEMBLY
TURBINE EXHAUST DUCT
FUEL PUMP
STANDPIPE
OXIDIZER 98.4 FT3 VOLUME
1400°F GAS TEMP.
B-132 TURBINE MAN. PRESS. NO. 2 0-120 PSIA
W.F. =13.9 LB/SEC AT 750 PSIA
1" FLEX
B-137 OXID. TANK TEMP. (-Z) -35° TO -170°F
PROPELLANT ISOLATION VALVE (FUEL)
FUEL FILTER 2
1/4" FUEL
GAS GENERATOR G.G. Pc = 485 PSIA
B-31 FUEL PUMP INLET TEMP. 0°/100°F
1/4" FLEX HOSE
W.F. 1.370 LB/SEC 1045 PSIA
B-1 FUEL PUMP INLET PRESS 0-100 PSIG
OVERBOARD DRAIN
3/8" FLEX HOSE
5/8" TUBE
1116 PSIA W.O. = 0.205 LB/SEC
CHECK VALVE AND BLEED PORT (OXIDIZER)
CHECK VALVE AND BLEED PORT (FUEL)
B-11 OX. VENTURI INLET PRESS. 0-1500 PSIG
GAS GENERATOR OXID. SOLENOID VALVE
GAS GENERATOR FUEL SOLENOID VALVE
N2 FILL VALVE OXIDIZER
OXIDIZER BLEED VALVE
OXIDIZER START TANK
START TANK TEMP. -300°F
OXIDIZER VENTURI 1
FUEL VENTURI 2
B-12 FUEL VENTURI INLET PRESS. 0-1500 PSIG
OXIDIZER FILL AND DRAIN VALVE

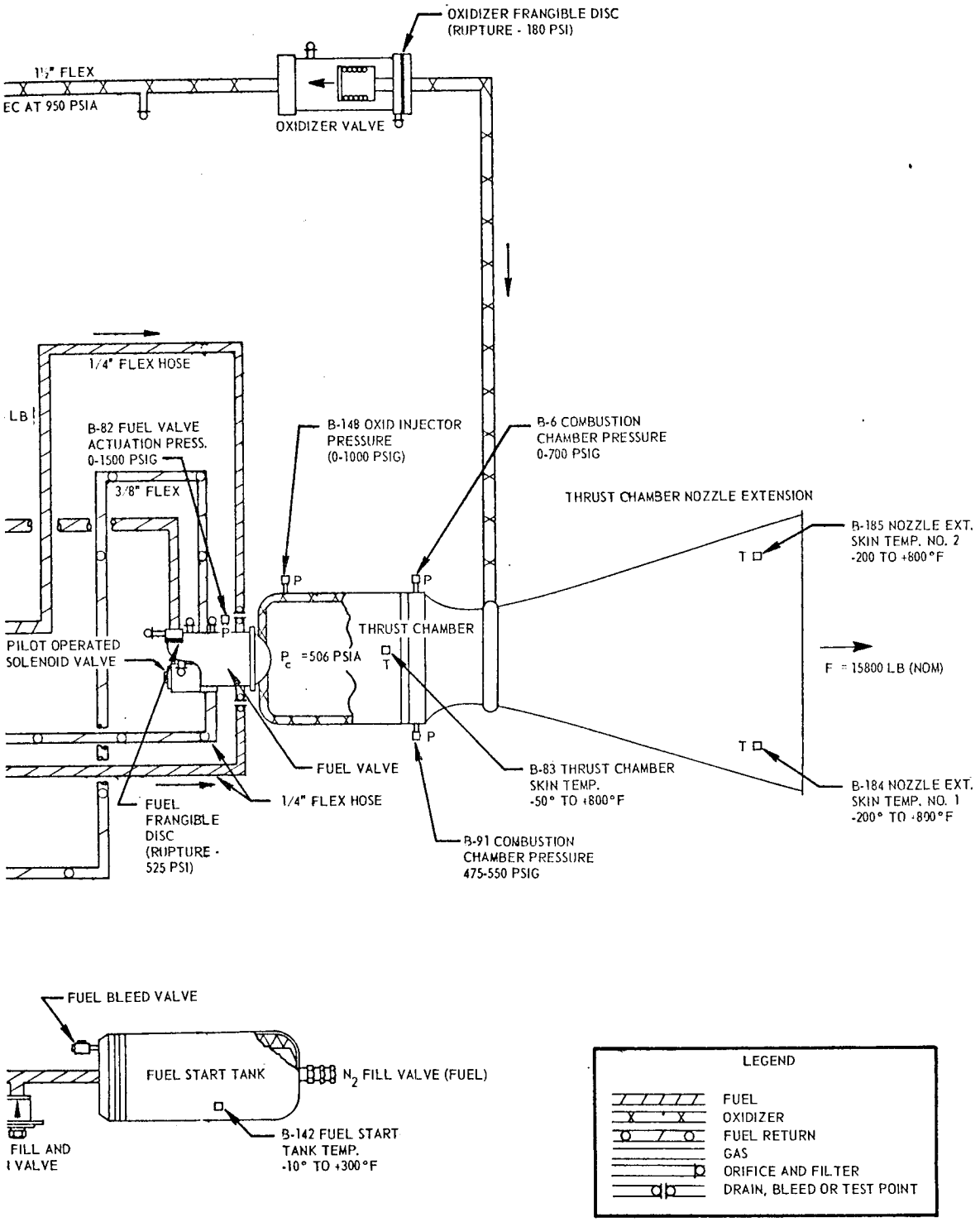


Fig. 2-2 Primary Propulsion System Fluid Flow Schematic

2.2 MAIN ENGINE

The XLR-81-BA-13 (Figs. 2-3, 2-4, and 2-5) is a liquid bi-propellant rocket engine with a minimum capability of five starts and a demonstrated capability of fifteen starts under vacuum conditions. This rocket engine consists of the following major components:

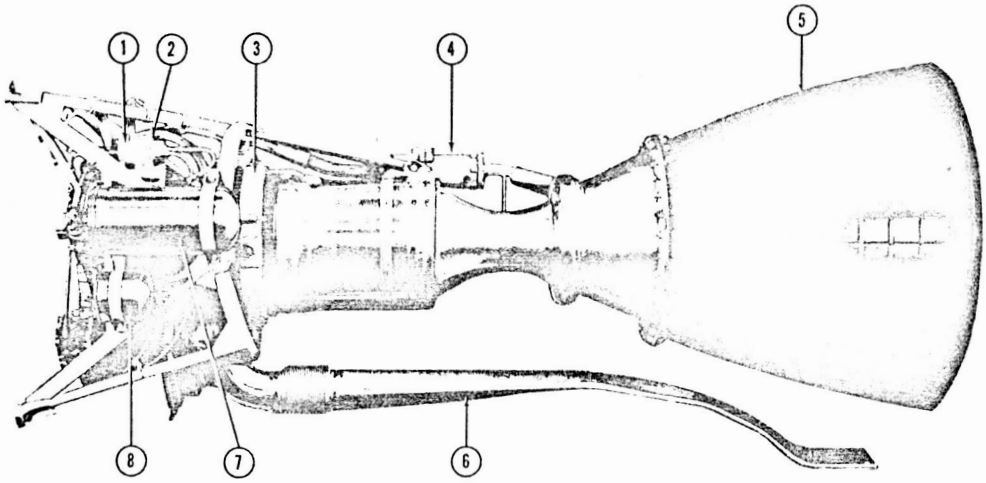
- Thrust-chamber assembly
- Multiple-restart assembly
- Turbine-pump assembly
- Overspeed shutdown electronic-gate and cable assembly
- Turbine-exhaust duct
- Propellant manifolds
- Thrust-chamber nozzle extension

Fuel used is unsymmetrical dimethylhydrazine (UDMH); oxidizer used is inhibited red fuming nitric acid (IRFNA). The propulsion system provides the thrust necessary to place the ATV into a selected orbit and to accomplish major orbital changes. A minimum of five starts is available for performing these maneuvers.

2.2.1 Electronic Gate

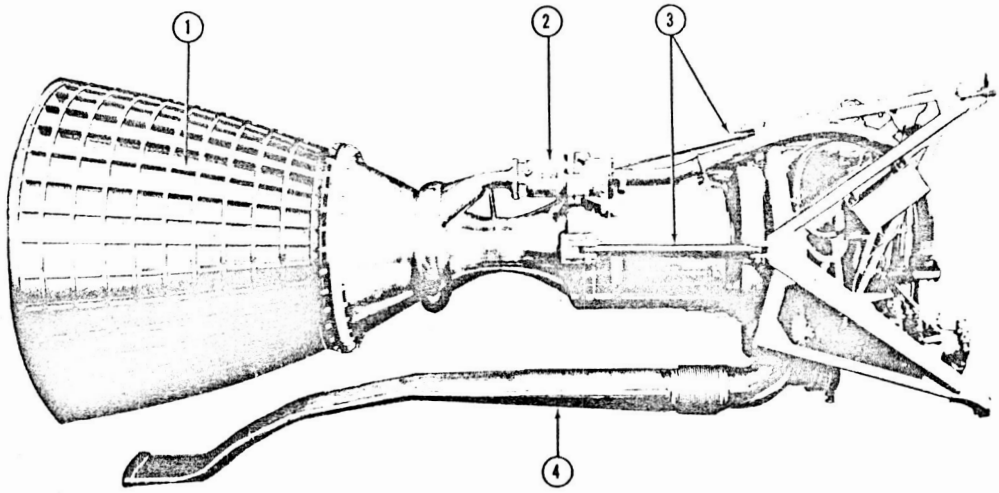
Purpose – The electronic gate (Fig. 2-6) serves a multiple purpose. During normal engine operation, this device is a junction and power-distribution box to the various engine solenoids. In the event of malfunction, it electronically senses turbine overspeed and takes immediate remedial action by removing power from the engine solenoid valves, thereby terminating engine operation.

Normal Engine Operation – The Model 8247 rocket engine commences operation any time that a voltage (nominally +28vdc) is applied to either pin A or D and a ground is applied to either pin B or E of J6000. When either power or ground is removed from these pins, engine shutdown occurs. Note from Fig. 2-5 that the start signal when applied at the aforementioned pins, goes (1)



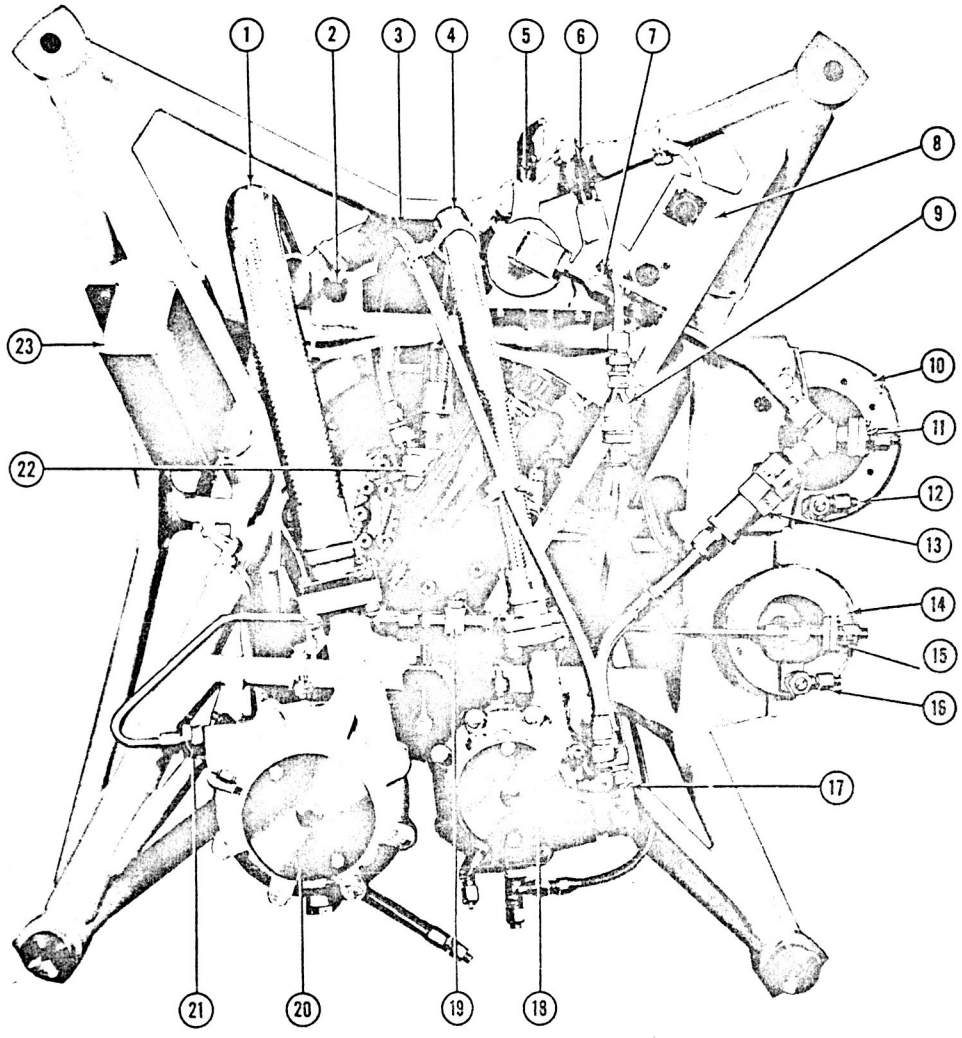
1 Gas Generator Oxidizer Solenoid Valve
2 Gas Generator Fuel Solenoid Valve
3 Gimbal Ring
4 Oxidizer Valve
5 Thrust Chamber Nozzle Extension
6 Turbine Exhaust Duct
7 Fuel Start Tank
8 Oxidizer Start Tank

Fig. 2-3a Rocket Engine – Left Side View



1 Thrust Chamber Nozzle Extension
2 Oxidizer Valve
3 Thrust Chamber Tie Bars
4 Turbine Exhaust Duct

Fig. 2-3b Rocket Engine – Right Side View



1 Oxidizer Manifold
2 Motional Pickup Connector
3 Fuel Valve Balance Return Line
4 Fuel Manifold
5 Gas Generator Fuel Solenoid Valve
6 Gas Generator Oxidizer Solenoid Valve
7 Fuel Venturi and Filter Assembly
8 Engine Electrical Interface Connectors
9 Oxidizer Venturi and Filter Assembly
10 Fuel Start Tank
11 Fuel Fill and Drain Valve
12 Fuel Start Tank Bleed Valve
13 Fuel Dual Check Valve
14 Oxidizer Start Tank
15 Oxidizer Fill and Drain Valve
16 Oxidizer Start Tank Bleed Valve
17 Fuel Filter
18 Fuel Pump
19 Oxidizer Dual Check Valve
20 Oxidizer Pump
21 Oxidizer Filter
22 Fuel Burst Disc Holder
23 Electronic Gate

Fig. 2-4 Rocket Engine — View Looking Aft

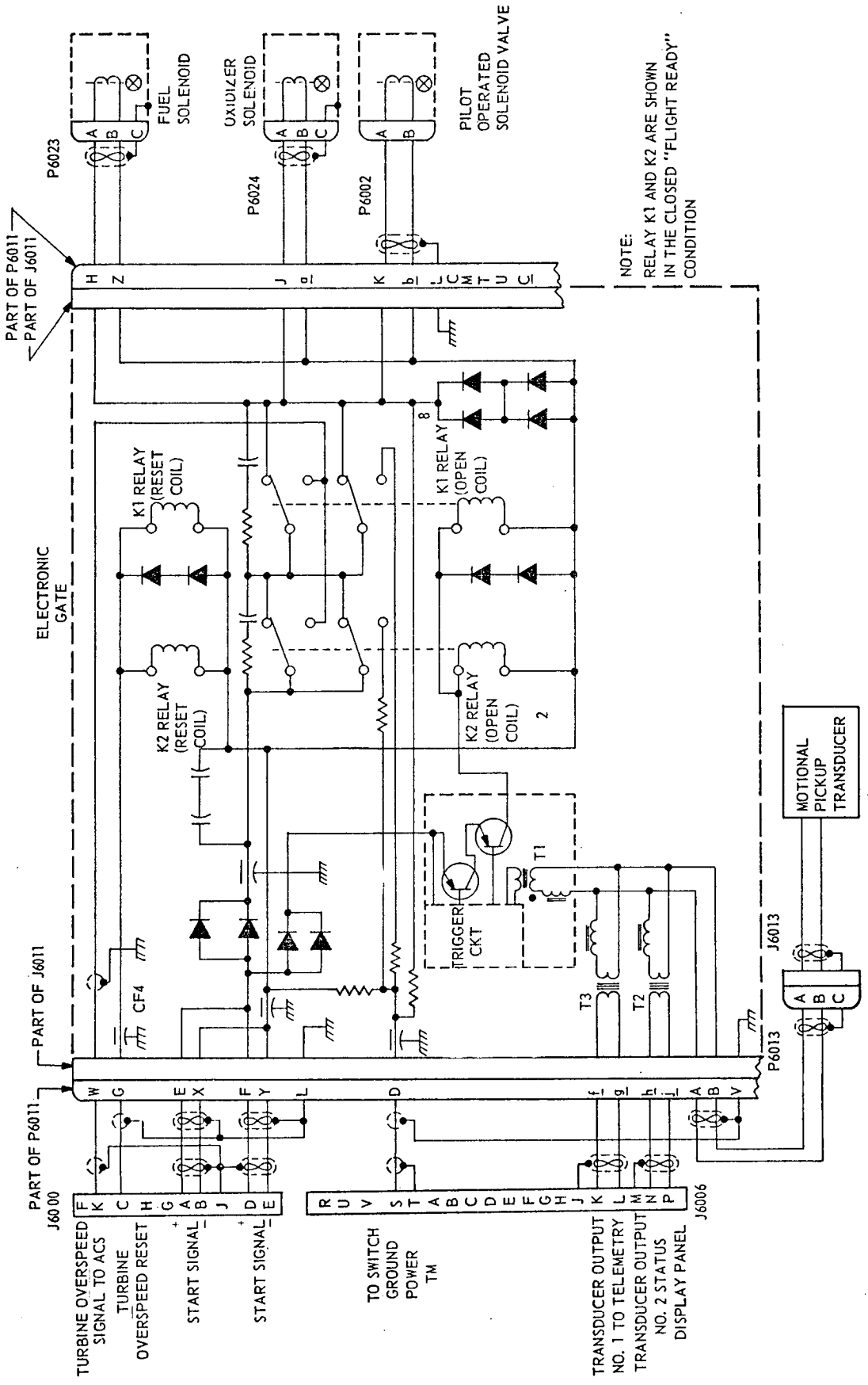


Fig. 2-5 Rocket Engine Electrical Schematic Diagram

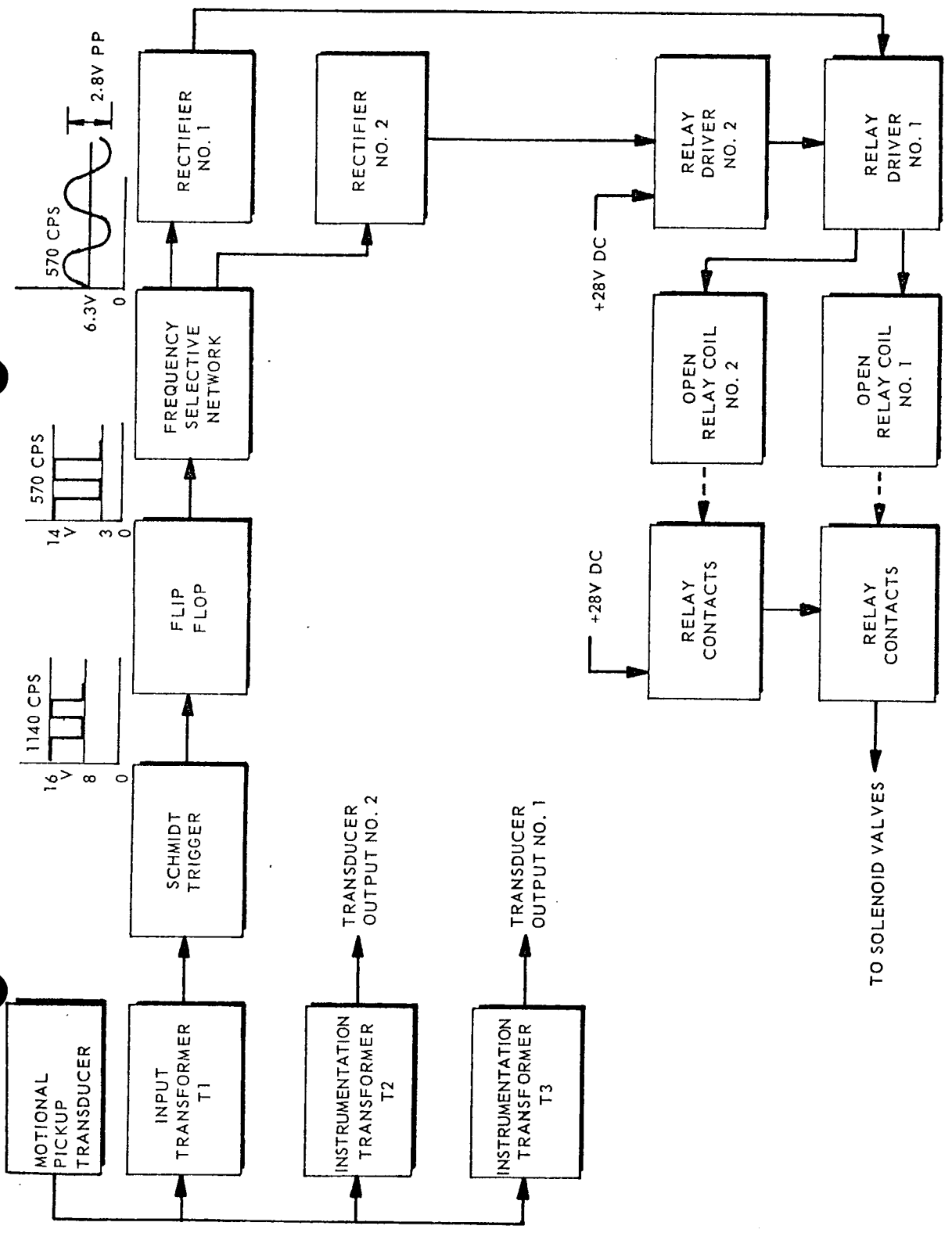


Fig. 2-6 Electronic Gate Overspeed Circuit Block Diagram

through parallel diodes which provide polarity reversal protection to the sensitive electric circuitry of the overspeed system, and series-parallel wired K-1 and K-2 relay contacts which distribute the signal to the gas generator fuel solenoid valve (GGFSV), the gas generator oxidizer solenoid valve (GGOSV), and finally the pilot-operated solenoid valve (POSV). Various capacitors, diodes, and resistors are used to suppress any radiated electromagnetic interference (EMI) to other electronic components in the ATV or spacecraft.

Engine Reset Technique – Prior to each intended operation of the Model 8247 engine, a short timed-pulse (approximately 50 μsec long) is sent, via a fixed capacitive-resistive type circuit, into pin C of J6000. The purpose of sending this pulse is to apply a brief resetting signal to both K-1 and K-2 relays. In case of turbine overspeed, this feature allows the resetting of the electronic gate whereupon another reset signal to obtain normal engine operation may be sent at the discretion of either a ground station or the spacecraft. This reset signal is sent and terminated prior to intended Model 8247 operation in order to preclude the possibility of having a reset signal holding the K-1 and K-2 contacts in the reset position at the same time that the engine is experiencing turbine overspeed and sending a K-1 and K-2 open signal to the same contacts.

Engine Overspeed Operation – The circuitry and operation of the turbine overspeed portion of the electronic gate is described by the block diagram (Fig. 2-6).

The turbopump gear-case-mounted, magnetically sensitive motion pickup transducer (Fig. 2-7) senses turbine speed by noting the variance in flux density as the four lightning holes in the oxidizer gear pass the pickup head. These holes vary the air gap and change the flux density as they rotate past the sensing head, thus causing the transducer to emit four pulses for each revolution of

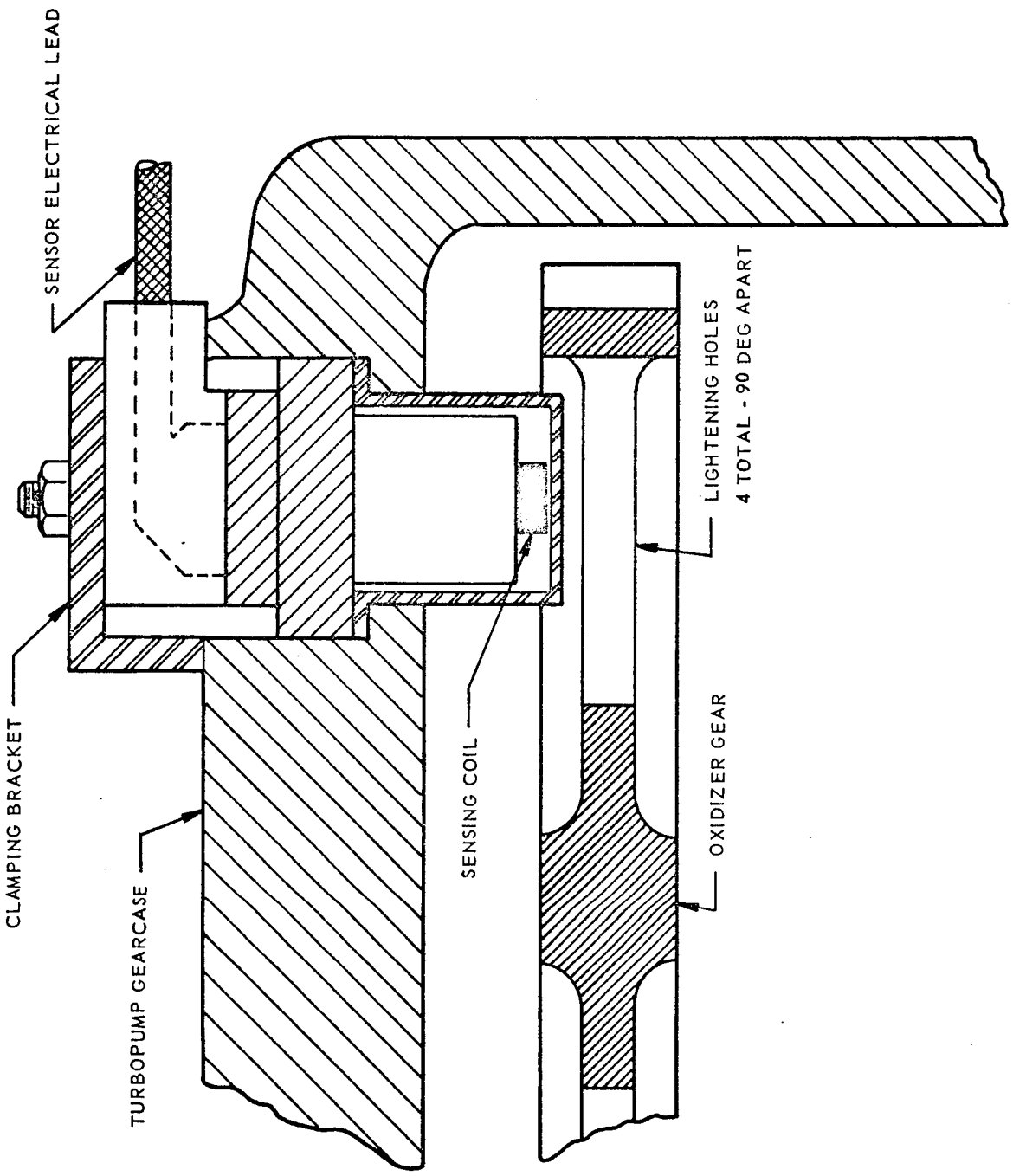


Fig. 2-7 Mounting of Motional Pickup Transducer

the oxidizer gear. The relationship of turbine speed (N_t) in revolutions per minute (rpm) to transducer output (f_{out}) in cycles per second (cps) is as follows:

(1) Ratio of turbine wheel to oxidizer gear is 74:43.

(2) $N_t = (f_{out})$ (cycles/sec) (74/43) (60 sec/min) (rev/cycle)

therefore,

$$N_t = (25.814)\,(f_{out})$$

(3) Thus, given a pulse rate of 1142 cps,

$$N_t = (1142)\,(25.814) = 29,480 \text{ rpm}$$

The transducer puts out a signal having an approximate sinusoidal wave form (Fig. 2-6), which at a frequency of 1140 cps is nominally 2.8 volts peak-to-peak (Vpp). This transducer output is fed into pins A and B of P6011 and through three isolation transformers (Figs. 2-5 and 2-6). Transformer T_1 is the input transformer for the electronic gate circuitry whereas T_2 and T_3 are output transformers to the ATV Status Display Panel and telemetry, respectively. These isolation transformers are used to preclude the possibility of a short circuit in vehicle wiring affecting the proper operation of the overspeed circuit.

Overspeed Signal Condition – The Schmidt trigger circuitry picks up the output signal from transformer T_1. When the input signal approaches 1140 ±20 cps (turbine overspeed condition), the voltage level becomes high enough to fire the Schmidt trigger. The input signal is then converted into a square wave, as shown in Fig. 2-6, and fed into the circuitry of the flip-flop. The flip-flop takes the square-wave input, cuts the frequency in half, and feeds this signal into the frequency-selective network.

This frequency-selective network consists of a band pass filter and wave-shaping circuitry. The band pass filter allows only a selected-signal frequency to pass through, thus cutting out any harmonics and allowing only the basic signal to pass. The square wave input is reshaped into a sine wave.

Then, the signal is fed into a pair of rectifiers and changed into a 2.0 vdc signal which drives the relay drivers in a mode that completes the circuit between the 28-vdc vehicle power supply and the "OPEN" relay coils. When these relay coils are energized, the relay contacts are tripped, thus breaking the circuit to the engine solenoid valves. Removing power from the valves allows them to close under the spring load, which terminates the flow of propellants to the gas generator and the thrust chamber.

2.2.2 Engine Start System

Purpose of System – A means of reliably starting the Model 8247 rocket engine for initial burn is required. Further restriction is placed upon the Model 8247 engine inasmuch as a minimum of four additional engine restarts must be available to complete the Gemini-ATV mission, and all of these starts must be accomplished under relative zero-g conditions. The previously used Model 8096 rocket engine accomplished start and restart by means of two solid-propellant charges mounted in containers which were integral part of the gas generator assembly. The obvious disadvantage of five or more such solid-propellant charges necessitated a new design concept which would have reliability, simplicity, and inherent multiple-restart capability.

Start Tank Operation – Thin-shell engine-mounted tanks were designed for containing both liquid propellants and high-pressure gas which are separated by a thin moveable-metal bellows. The propellant volume in each tank is sufficient to start combustion in the gas generator and maintain this combustion until the turbopump assembly attains sufficient discharge pressure to take over the supply of propellant to the gas generator, as well as

replenish the propellant removed from the tank to accomplish the starting phase. The gas side of the tank is designed to be filled with an inert gas (nitrogen) at a sufficiently high pressure to force the propellant through the interconnecting plumbing into the gas generator at approximate nominal gas generator supply pressures. Once filled, this gas remains trapped and reduces in pressure only when propellants are demanded from the start tanks during the start phase of engine operation. Then the metal bellows moves, forcing liquid from the tank, and thereby increasing the gas ullage space. As the turbopump assembly accelerates and produces propellant-discharge pressures greater than the pressure then existing on the gas side of the start tank, the bellows reverses its direction of movement and the tank starts to refill with liquid, while at the same time the gas side pressure rises until equilibrium is reached as the gas side pressure approximately equals the pump-discharge pressure. The start tank is thusly replenished for another restart operation upon later demand. Examination of the engine schematic (Fig. 2-2) reveals that upon engine shutdown, the liquid propellants are trapped between the dual check valves and the closed gas generator solenoid valves.

Start Tank Material – The tank shells are machined from a type 355 stainless-steel closed die forging. Prior to final machining, these tanks are heat treated to SCT (sub-zero cooling temperature - 185,000 to 205,000 psi tensile).

2.2.2.1 Fuel Start Tank. A description of the engine start system fuel tank follows.

Tank Volumes and Pressures – The fuel start tank (Fig. 2-8) has a total nominal volume of 187.9 $in.^3$ of which 95 $in.^3$ is the maximum expellable, 7.6 $in.^3$ is trapped residual (i.e., will not come out with normal draining), and 85.3 $in.^3$ is gas ullage space. Maximum allowed operating pressure is 1875 psig; nominal tank working pressure is approximately 980 psig.

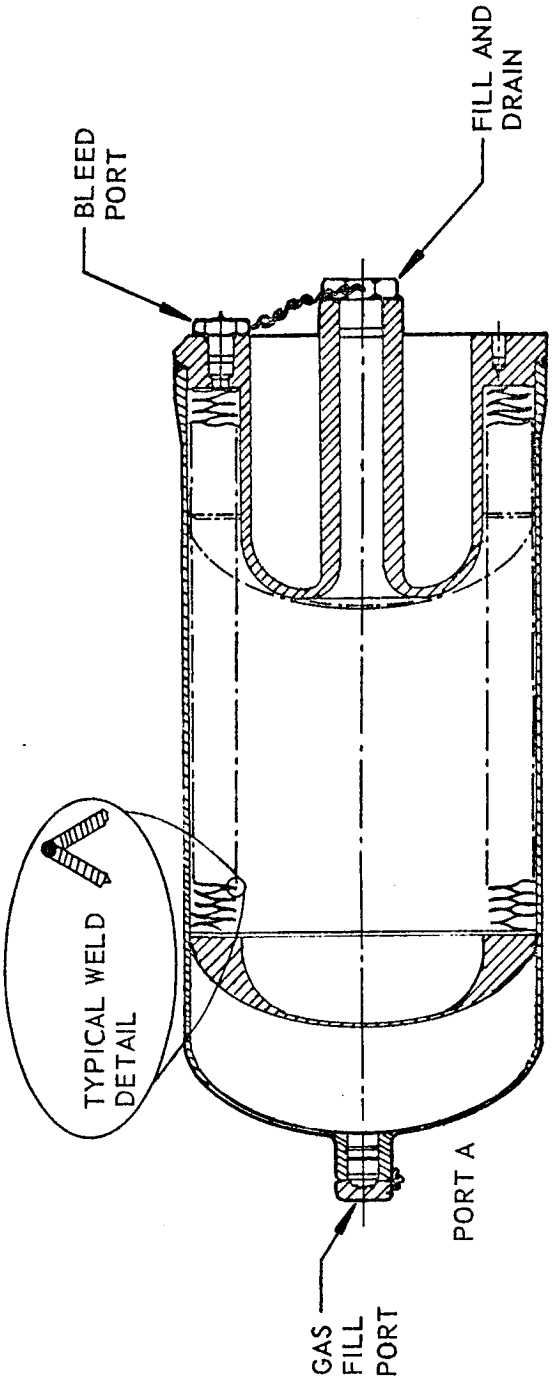


Fig. 2-8 Fuel Start Tank

Tank design burst pressure is approximately four times the nominal working pressure; therefore, personnel can safely work in the immediate area while the tank is being pressurized.

Tank Construction – The bellows is a nested ripple-type convolution design. When suitably restrained in the longitudinal and lateral directions and extended to 10 in., the bellows withstands a 100-psid internal pressure differential; when compressed to 1.960 in., it withstands 800-psid external pressure differential. The bellows is capable of withstanding a minimum of 50,000 cycles (each cycle being an extension and compression as previously described). The material used is AM 350 corrosion-resistant steel. Using the automatic Tungsten Inert Gas process, the bellows is constructed by welding at the convolutions with gas shielding on both sides of the welds. Each bellows consists of 65 convolutions. The final bellows assembly is leak tight when pressurized with helium to 50 psid internally and extended to 10 in. and restrained suitably in the longitudinal and lateral directions. During normal tank operation, the pressure differential across the bellows is negligible.

Auxiliary Tank Valves – Three valves provide access to the start tank. They are:

1. The manually operated fuel bleed valve (Fig. 2-9) which provides a means of flushing and functionally checking the start tank. This valve has a single, hand-operated, metal-to-metal sealing valve stem. The bleed port and the valve actuator are covered by an AN-919 pressure cap at all times other than during checkouts or loading, thereby providing a redundant pressure seal.

2. The N_2-gas fill valve (Fig. 2-10) which is used to pressurize the gas side of the start tank. It is a military standard part, having a single poppet with a metal-to-metal seal. An AN-919 pressure cap is used as a redundant pressure seal to insure that leakage does not occur.

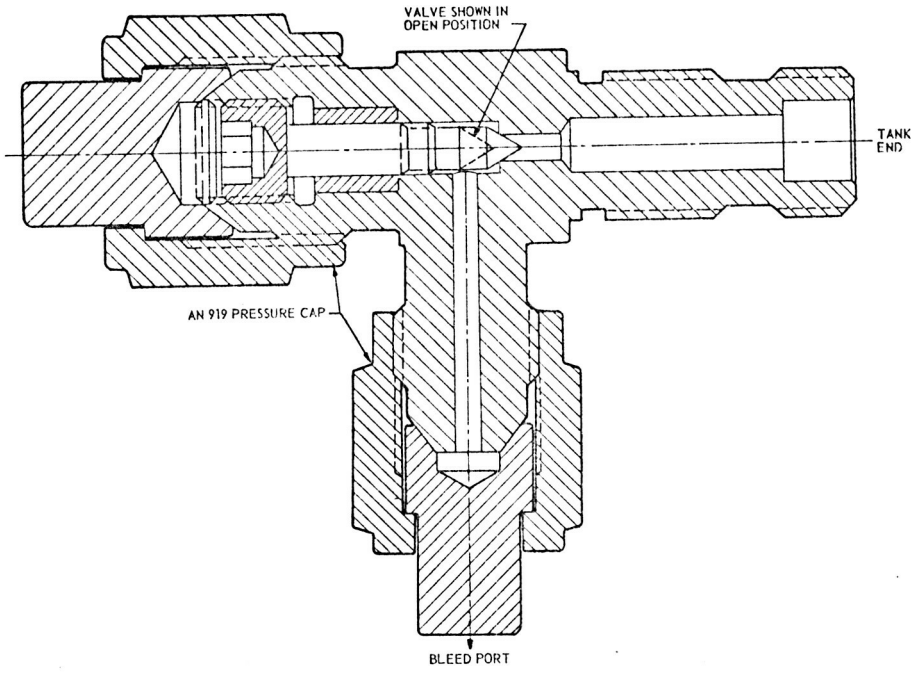


Fig. 2-9 Fuel Start Tank Manual Fuel Bleed Valve

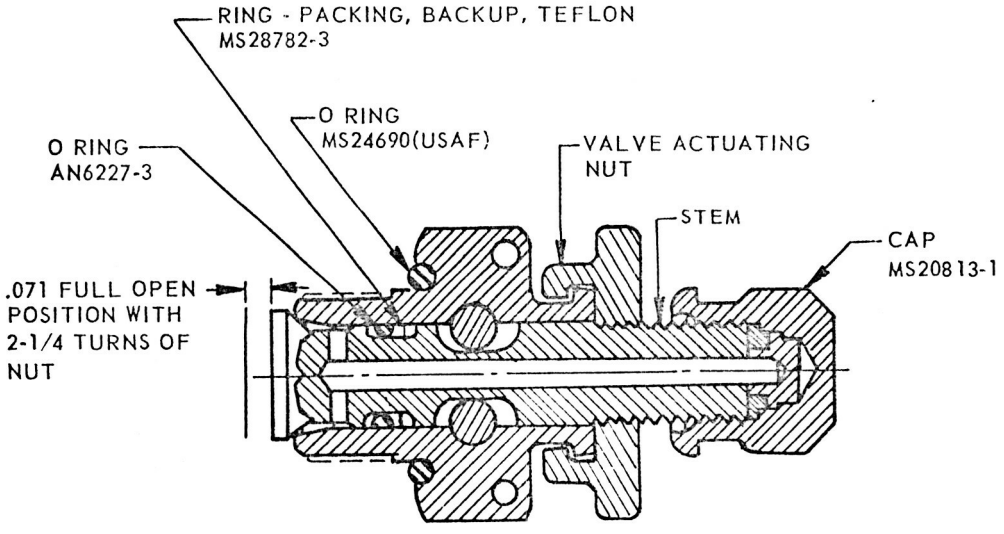


Fig. 2-10 High-Pressure Nitrogen Charging Valve (Start Tank)

3. The fill and drain check valve (Fig. 2-11) which accomplishes initial loading of the start tanks. Fill is accomplished through the single-poppet check portion of the valve. The remainder of the fluid passages are open, serving as a junction for lines coming from the fuel pump and the start tank and going to the gas generator. The fill port is capped with an AN-919 pressure cap to provide a redundant seal at all times other than during tank loading and checkout.

2.2.2.2 Oxidizer Start Tank. A description of the engine start system oxidizer tank follows.

Tank Volumes and Pressures — The oxidizer start tank (Fig. 2-12) has a total nominal volume of 31 in.^{3} of which 13.8 in.^{3} is the maximum expellable, 2.2 in.^{3} is trapped residual (i.e., will not come out with normal draining), and 15.0 in.^{3} is gas ullage space. Maximum allowable operating pressure is 1770 psig; nominal tank working pressure is approximately 975 psig. Tank design burst pressure is approximately four times the nominal working pressure; therefore, personnel can safely work in the immediate area while the tank is being pressurized.

Tank Construction — The bellows is of nested ripple-type convolution design. When suitably restrained in the longitudinal and lateral directions and extended to 3 in., the bellows withstands a 100-psid internal pressure differential when compressed to 0.970 in., it withstands 800-psid external pressure differential. The bellows is capable of withstanding a minimum of 50,000 cycles (each cycle being an extension and compression as previously described). The material used is type 347 stainless steel. Using the automatic Tungsten Inert Gas process, the bellows is constructed by welding at the convolutions with gas shielding on both sides of the welds. Each bellows consists of 32 convolutions. The final bellows assembly is leak tight when pressurized with helium to 50 psid internally and extended to 3 in. and restrained suitably in the longitudinal and lateral directions.

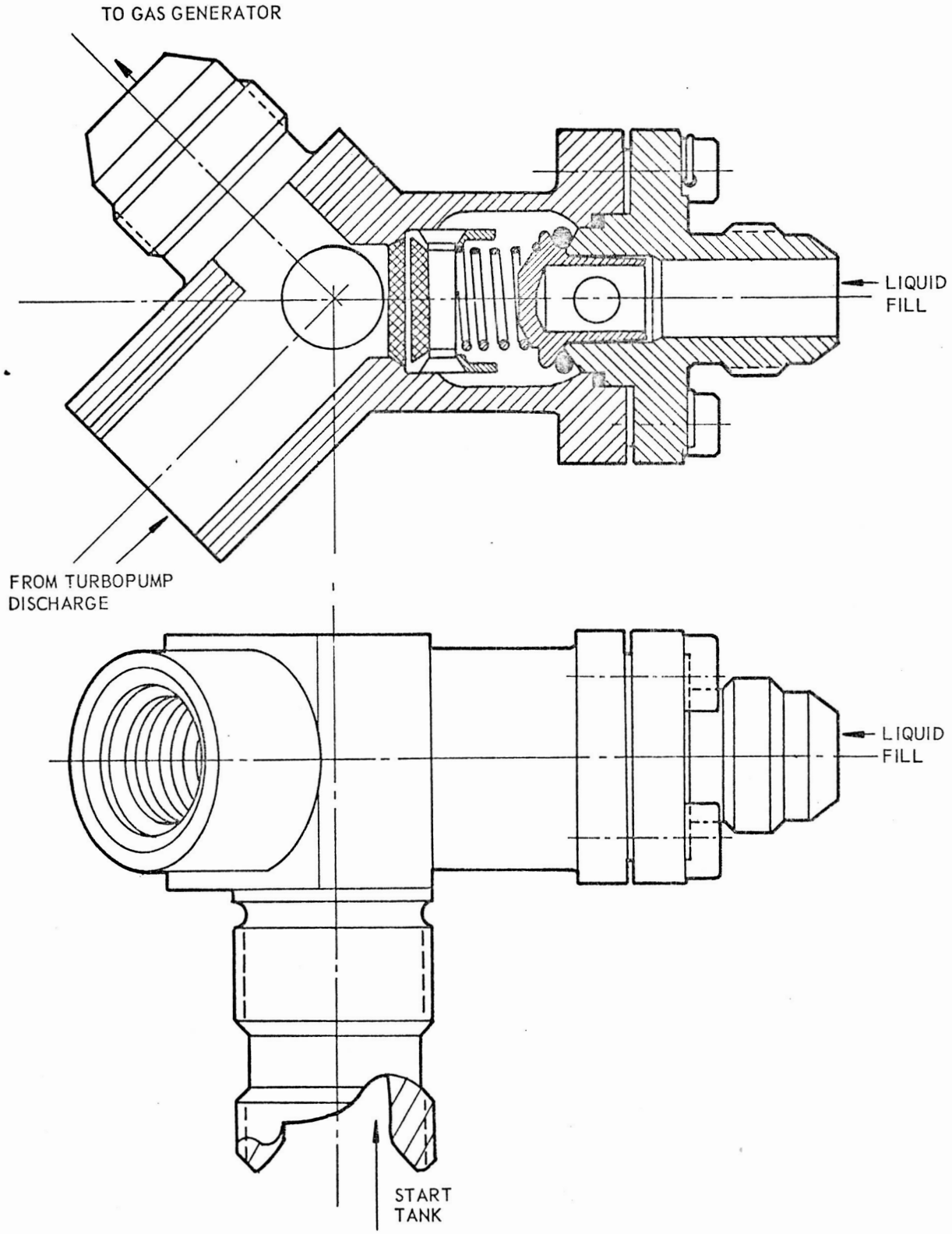


Fig. 2-11 Fuel Start Tank Fill and Drain Valve

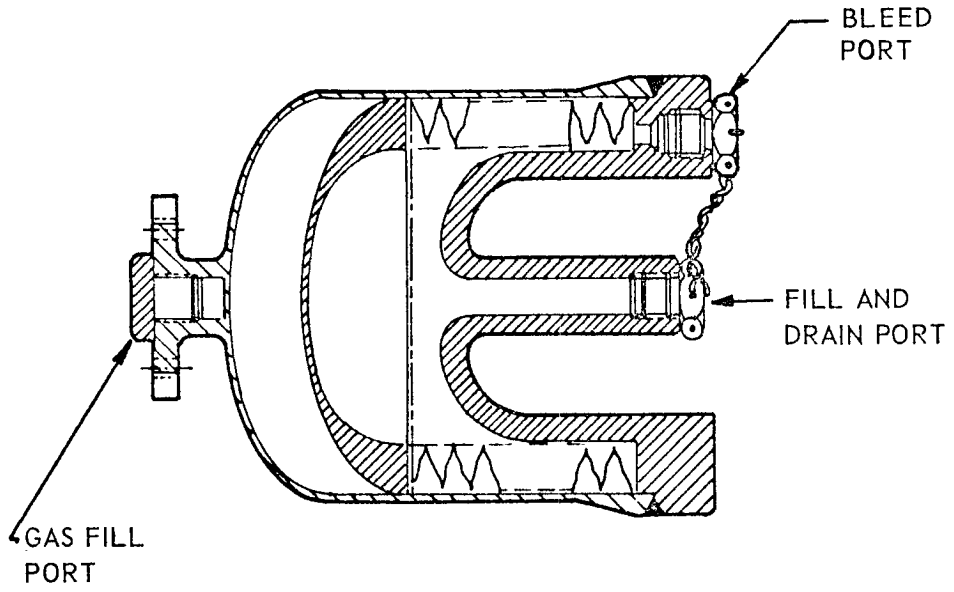


Fig. 2-12 Oxidizer Start Tank

Auxiliary Start Tank Valves – Three valves provide access to the start tanks. They are:

(1) The oxidizer bleed valve which is identical to the fuel bleed valve (Fig. 2-9) except for thread sizes.

(2) The N_2-gas fill valve which is identical to the fuel start-tank gas-fill valve (Fig. 2-10).

(3) The fill and drain check valve (Fig. 2-13) which accomplishes initial loading of the start tank. Fill is accomplished through the single-poppet check portion of the valve. The fluid passages are open to the start tank and the oxidizer pump. The fill port is capped with an AN-919 pressure cap to provide a redundant seal at all times other than during tank loading and checkout.

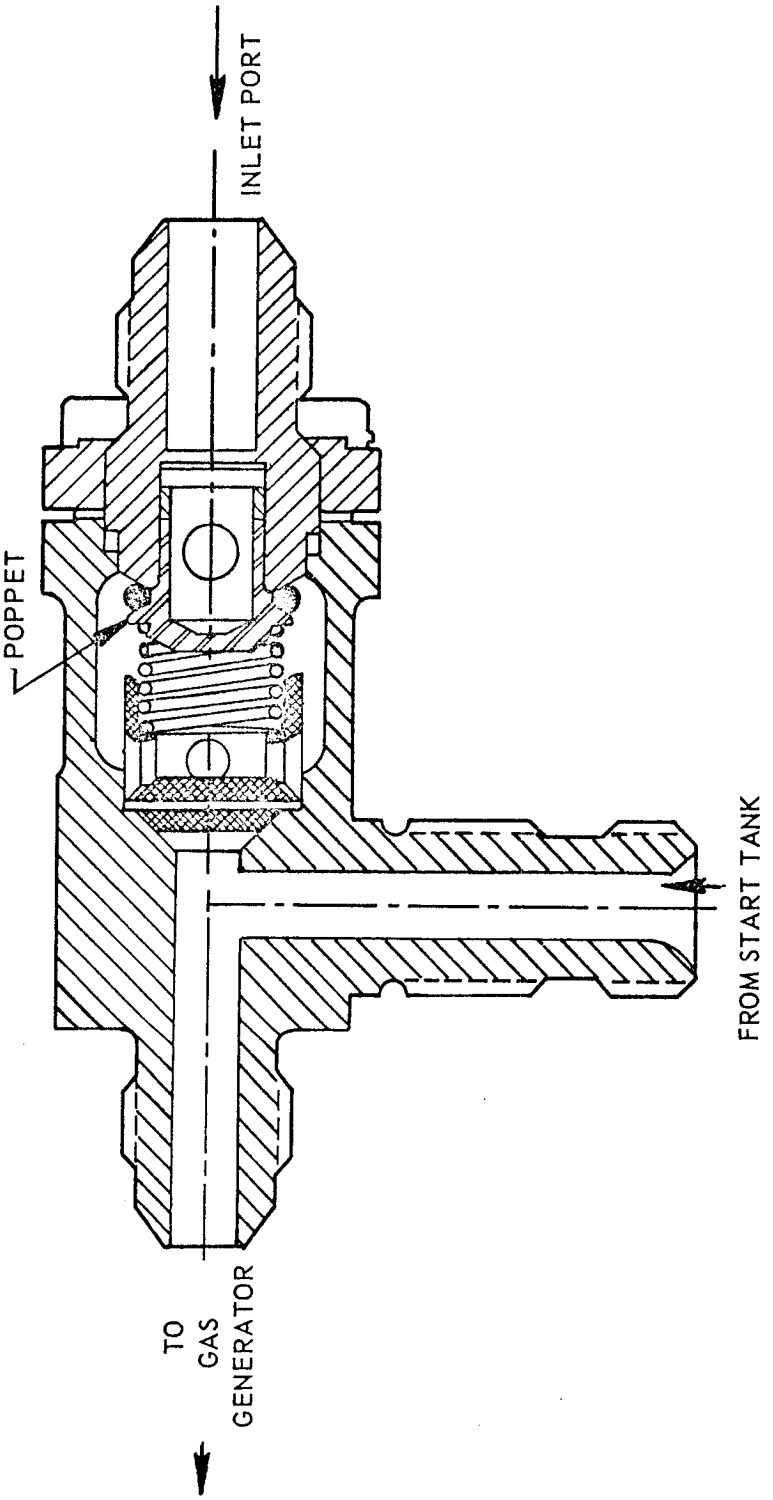


Fig. 2-13 Oxidizer Start Tank Fill and Drain Valve

2.2.3 Cavitating Venturis

Purpose – The cavitating venturis provide primary control of engine thrust level and are accurately calibrated to regulate flow of oxidizer and fuel to the gas generator. Since the gas generator provides power for driving the turbine pumps at the speed required to supply the thrust chamber at rated flow, the venturis indirectly control the engine thrust level by controlling the amount of propellant to reach the gas generator.

Venturi Construction – Figure 2-14 shows a typical venturi cross-section. Fuel and oxidizer venturis are similar except for size. Venturis are built of stainless steel for corrosion and erosion resistance and are machined internally to an extremely smooth finish for high efficiency. The converging section is short and the diverging section is relatively long. Throat diameters vary but are approximately 0.031-in. in diameter for the oxidizer and 0.106-in. in diameter for the fuel. They have an integrally mounted 165-micron filter.

2.2.4 Dual Check Valves, Fuel and Oxidizer

Purpose – The dual-poppet check valves (Fig. 2-2) are an integral part of the start system. Prior to each engine burn, the start system is maintained in a pressurized condition in preparation for the next engine start. The purpose of these check valves is to maintain under pressure the propellant trapped between the start tank, gas generator solenoid valves, and the turbopump discharge port (bootstrap port). This dual-poppet design was selected to provide a redundant seal and thus increase reliability of the system.

Engine Operation – At the initiation of the engine-start signal, the check valves are closed, thus containing the propellants within the start system. As fuel pump and oxidizer pump bootstrap pressure increase above the decaying (blowdown) fuel and oxidizer start tank pressures, the dual check valves open and allow propellant flow to sustain gas generator operation and recharge the fuel and oxidizer start tanks.

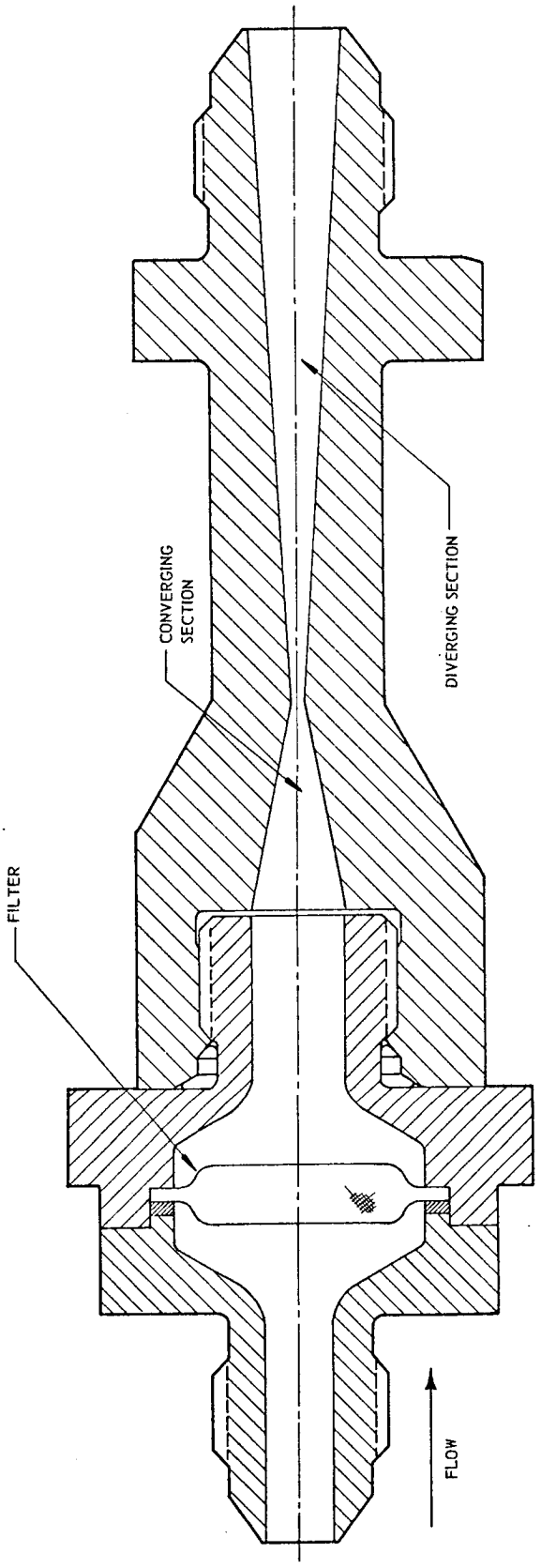


Fig. 2-14 Fuel and Oxidizer Venturi and Filter Assembly

2.2.4.1 Fuel Dual Check Valve. A description of the fuel dual check valve follows.

Design—The fuel dual check valve (Fig. 2-15) body and poppets are constructed of 2024 T-4 aluminum alloy. The retaining spring is fabricated of passivated type 316 stainless steel and is designed to impart a spring force such that upstream off-seating pressure is 5 psig maximum. Dynamic seals are virgin TFE teflon on metal seats. Static seals are butyl rubber 805-70 compound. The unit is designed to have a working pressure of 1875 psig, a proof pressure of 2812 psig, and a burst pressure in excess of 3750 psig. Maximum pressure drop through the valve at working pressure is 40 psid.

Checkout—A checkout port is provided in the body and between the two poppets for pre-test checkout. This port is pressure capped during normal engine operation. Each poppet has a liquid leakage rate of less than 0.10 cc in 24 hours when under normal working pressures.

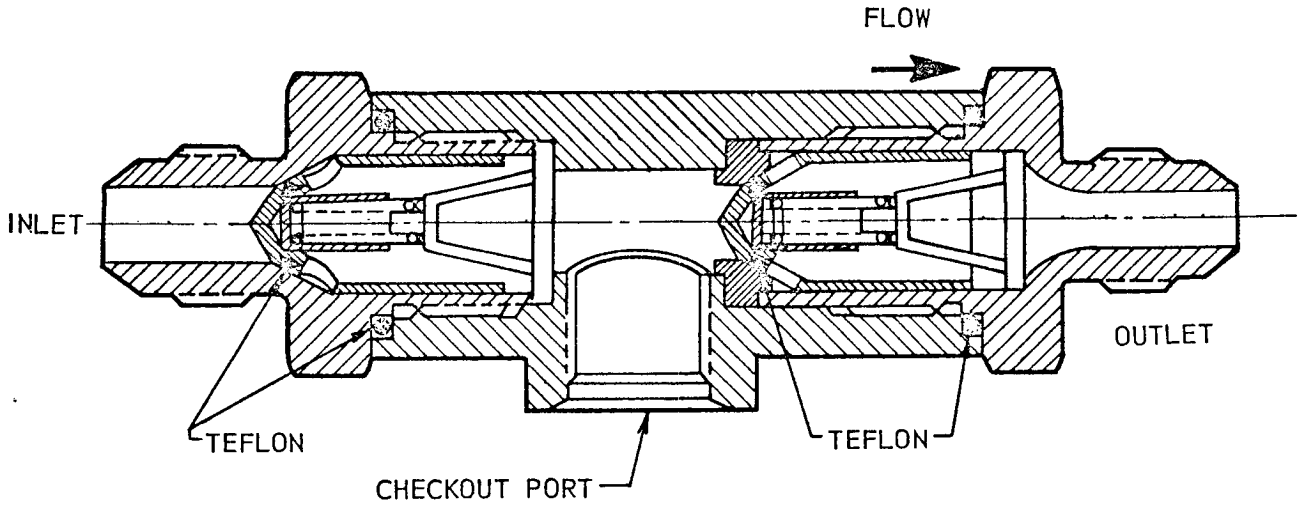


Fig. 2-15 Fuel Dual Check Valve

2.2.4.2 Oxidizer Dual Check Valve. A description of the oxidizer dual check valve follows.

Design—The oxidizer dual check valve (Fig. 2-16) body and poppets are constructed of passivated type 304 stainless steel. The retaining spring is fabricated of passivated type 316 stainless steel and is designed to impart a spring force such that upstream off-seating pressure is 15 psig maximum. The poppet seals are virgin TFE teflon O-rings which butt against metal seats. In addition, the start tank side has a teflon seal between the seat and valve body. Static seals are of teflon. The unit is designed to have a working pressure of 1770 psig, a proof pressure of 2655 psig, and a burst pressure in excess of 3540 psig. Maximum pressure drop through the valve at working pressure is 30 psid.

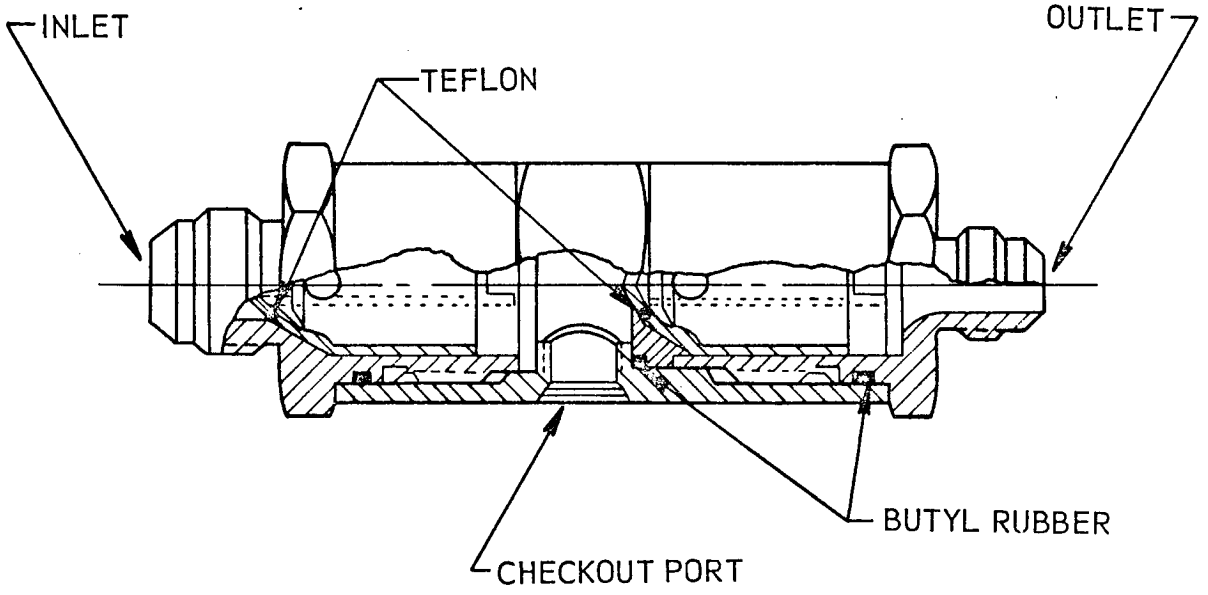


Fig. 2-16 Oxidizer Dual Check Valve

Checkout – A checkout port is provided in the body and between the two poppets for pre-test checkout. This port is pressure capped during normal engine operation. Each poppet has a liquid leakage rate of less than 0.10 cc in 24 hours when under normal working pressures.

2.2.5 Gas Generator Solenoid Valves

Purpose – The gas generator solenoid valves are spring-loaded normally closed electrically actuated valves, which, when open allow flow of propellants to the gas generator. These valves are maintained in an electrically energized open condition throughout the engine burn. Engine shutdown is initiated by removing power from the GG solenoid valves, which close under spring and dynamic loads and shut off propellant flow to the gas generator.

2.2.5.1 Gas Generator Fuel Solenoid Valve. A description of the gas generator fuel solenoid valve (GGFSV) follows. Refer to Fig. 2-17. Arrows show propellant flow passages; letters indicate the following:

A. Stainless Steel Liner – encapsulates the fluid passages to prevent any internal fuel (UMDH) leakage
B. Spring-Loaded Poppet Assembly – controls fuel flow to the gas generator; also a part of the poppet assembly is a ferrite core for electrical actuation
C. Seat Assembly – retains the teflon seat and spring which reduces shock on closure
D. Teflon Seat – provides the actual seal to shut off the oxidizer flow to the gas generator
E. Actuation Gap – indicates the amount of travel available for poppet actuation.

Actuation occurs when the engine-start signal is applied. The solenoid coils are energized, and the poppet retracts the amount at (E). This opens up a fluid passage between the seat assembly (C), the seat (D), and the poppet assembly (B) – allowing fuel to flow from the start tank into the gas generator.

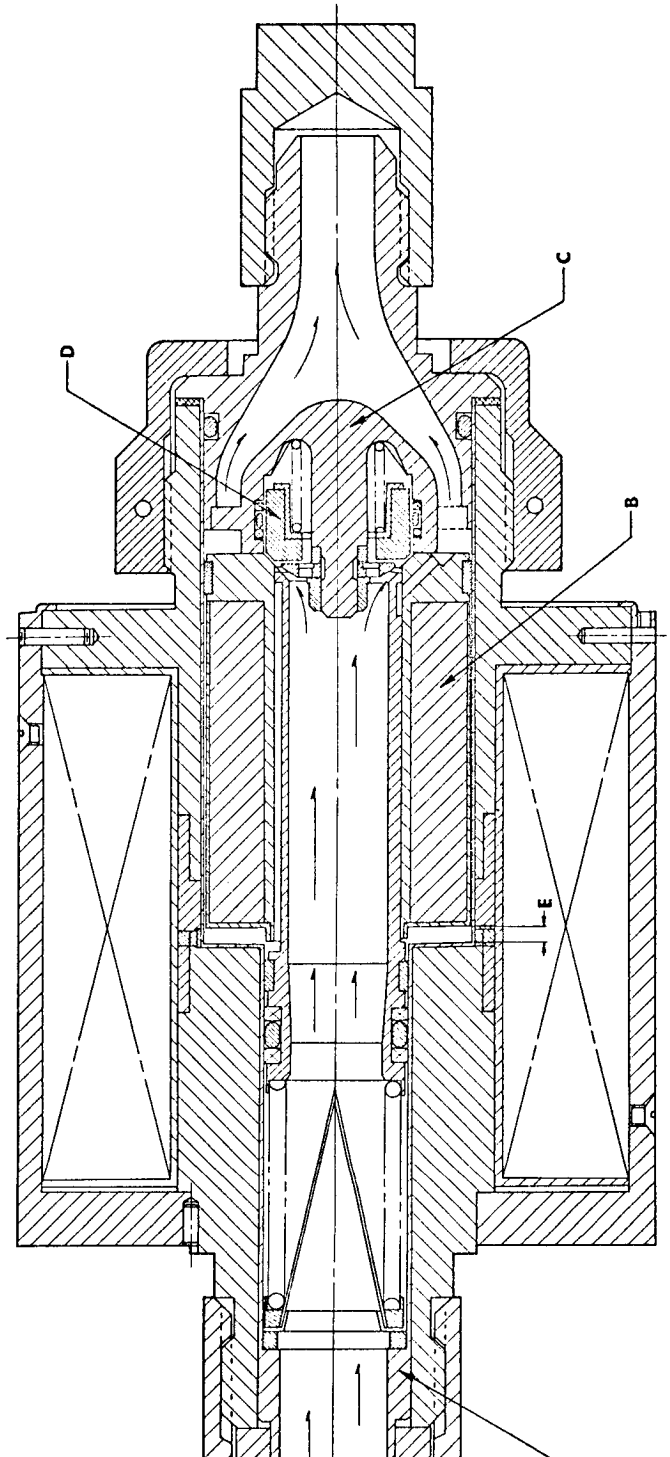


Fig. 2-17 Gas Generator Fuel Solenoid Valve

The valve remains open as long as the engine-start signal is applied. When the start signal is removed, electrical power is removed from the solenoid coils – allowing the poppet to close under spring and dynamic load and shutting off flow of fuel to the gas generator. The seat assembly (C) contains a spring to reduce closing shock on the teflon seat.

Table 2-1 gives pertinent design characteristics of the GGFSV.

Table 2-1

DESIGN CHARACTERISTICS OF GAS GENERATOR FUEL SOLENOID VALVE

1.	Pressure	Operating	1875 psig
		Proof	2812 psig
		Burst	3750 psig
2.	Temperature	Operating	-35°to +160°F
		Storage	-65° to +165°F
3.	Solenoid	Operating Voltage	18 to 30.5 vdc
		Current	3.15 amp at 30.5 vdc
		Resistance	9.7Ω at 70°F
4.	Response Time	Opening	0.100 sec max. at 18.0 vdc
		Closing	0.025 sec max. at 30.5 vdc
5.	Duty Cycle	Gas or Liquid Flow	Unlimited "on" at 30.5 vdc
		No Flow	10 min "on" at 30.5 vdc
6.	Flow		1.37 +0.07 lb/sec
7.	Leakage	Internal at 1875 psig	0.10 cc/24 hr max.
8.	Weight		3.4 lb max.
9.	Cycle life		6000 cycles

2.2.5.2 Gas Generator Oxidizer Solenoid Valve. A description of the gas generator oxidizer solenoid valve (GGOSV) follows. Refer to Fig. 2-18. Arrows show propellant flow passages; letters indicate the following:

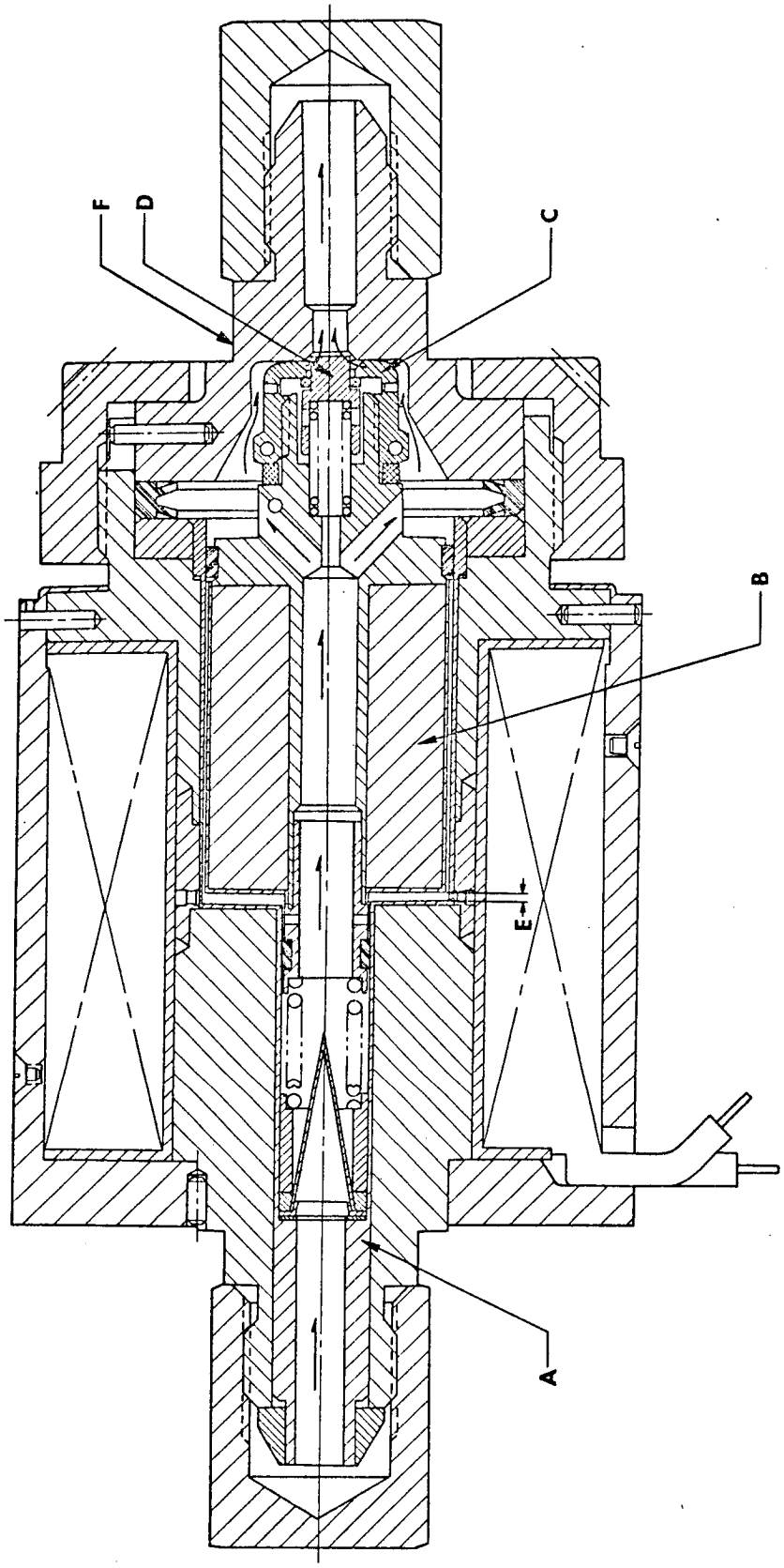


Fig. 2-18 Gas Generator Oxidizer Solenoid Valve

A. Stainless Steel Liner – encapulates the fluid passages to prevent any internal oxidizer (IRFNA) leakage

B. Spring-Loaded Poppet Assembly – controls oxidizer flow to the gas generator; also a part of the poppet assembly is a ferrite core for electrical actuation

C. Seat Retainer – a part of the poppet assembly, retains the Kel-F seat (D)

D. Kel-F Seat – provides the actual seal to shut off the oxidizer flow, has a spring load on it in addition to the main-poppet spring load

E. Actuation Gap – indicates the amount of travel available for poppet actuation (provides fluid opening of 0.022 to 0.024 in.).

Actuation occurs when the engine-start signal is applied. The coils are energized, and the poppet retracts the amount at (E). This opens up a fluid passage between the retainer seat (C), the seat (D), and the outlet fitting (F)– allowing oxidizer to flow from the start tank into the gas generator. The valve remains open as long as the engine-start signal is applied. When the start signal is removed, electrical power is removed from the solenoid coils – allowing the poppet to close under spring and dynamic load and shutting off flow of oxidizer to the gas generator.

For pertinent design characteristics refer to Table 2-2.

Table 2-2

DESIGN CHARACTERISTICS OF GAS GENERATOR OXIDIZER SOLENOID VALVE

1.	Pressure	Operating	1770 psig
		Proof	2655 psig
		Burst	3540 psig
2.	Temperature	Operating	-35° to +160°F
		Storage	-65° to +165°F
3.	Solenoid	Operating Voltage	18.0 to 30.5 vdc
		Current	1.7 amp at 30.5 vdc
		Resistance	18 to 20Ω at 70 F

Table 2-2 (Continued)

4.	Response Time	Opening	0.100 sec max. at 18.0 vdc
		Closing	0.025 sec max. at 30.5 vdc
5.	Duty Cycle	Gas or Liquid Flow	Unlimited "on" at 30.5 vdc
		No Flow	30 min "on" at 30.5 vdc
6.	Flow		0.205 lb/sec at 30 psi max. pressure drop
7.	Leakage	Internal at 1770 psig with operating fluid	0.10 cc/24 hr
8.	Weight		1.8 lb
9.	Cycle Life		6000 cycles

2.2.6 Gas Generator

Combustion products of the gas generator (GG) drive a single-stage turbine which operates the propellant pumps by means of a gear drive. Initial combustion in the GG is initiated by hypergolic ignition of propellants stored in the start tanks. These start tanks supply a sufficient quantity of propellants to support combustion in the gas generator until turbopump bootstrap takes place and flow from the propellant pumps reaches the GG combustion chamber.

The gas generator combustion chamber is made of stainless steel and is an integral part of the turbine inlet-manifold. A highly fuel-rich mixture of 0.150 lb of oxidizer per lb of fuel is used to maintain relatively low gas temperature to the turbine. Since the combustion temperature is relatively low (approximately 1400°F), regenerative cooling is not necessary for the gas generator combustion chamber. Nominal flow is 1.575 lb/sec (1.370 lb/sec of fuel and 0.205 lb/sec of oxidizer). The nominal chamber pressure is 485 psia.

2.2.7 Turbopump Assembly

The turbopump assembly (Fig. 2-19) consists of a single-stage inpulse-type turbine, a fuel pump and an oxidizer pump that are both gear-coupled to the turbine shaft, and a gear housing that serves as the assembly frame for the three major components (Figs. 2-20 and 2-21). Drain cavities between the primary and secondary seals are vented overboard. The oxidizer pump is sealed from the gear case by a sliding-ring-type primary seal, a double-lip secondary seal, and an ambient vented-drain cavity between both seals. To provide maximum protection against the possibility of acid leaking past the primary seal into the gear case, low-pressure nitrogen - Freon gas from the attitude control system gas bottles (2-14 psig) — is applied between the primary and secondary lip seals any time that oxidizer is in the engine.

2.2.7.1 Turbine. A description of the turbine follows.

Design and Operation — the turbine is a single-stage impulse-type designed to operate nominally at 24,800 rpm when driven by hot gases from a bi-propellant gas generator. The turbine inlet-manifold incorporates the gas generator chamber, adapting to the gas generator injector.

Exhaust gases are ducted overboard through the elliptical exhaust duct which accounts for approximately 200 lb of engine thrust. Although circular exhaust ducts on Model 8096 engines are scheduled for incorporation on Agena D serial number 92 and up, the Model 8247 engine will continue to use the elliptical exhaust ducts. The reason for changing to the round exhaust duct is that unsymmetrical flow patterns in the elliptical exhaust duct sets up disturbing vehicle torques which create a 7.5-lb control-gas penalty in correction of these torques. The round duct is also 2 lb lighter. However, the circular duct presents interference problems with the ATV program peculiar aft rack, and redesign of the aft rack is not considered feasible. Since the Gemini-ATV Attitude Control System gas supply is not critical, the extra requirement presents no problem.

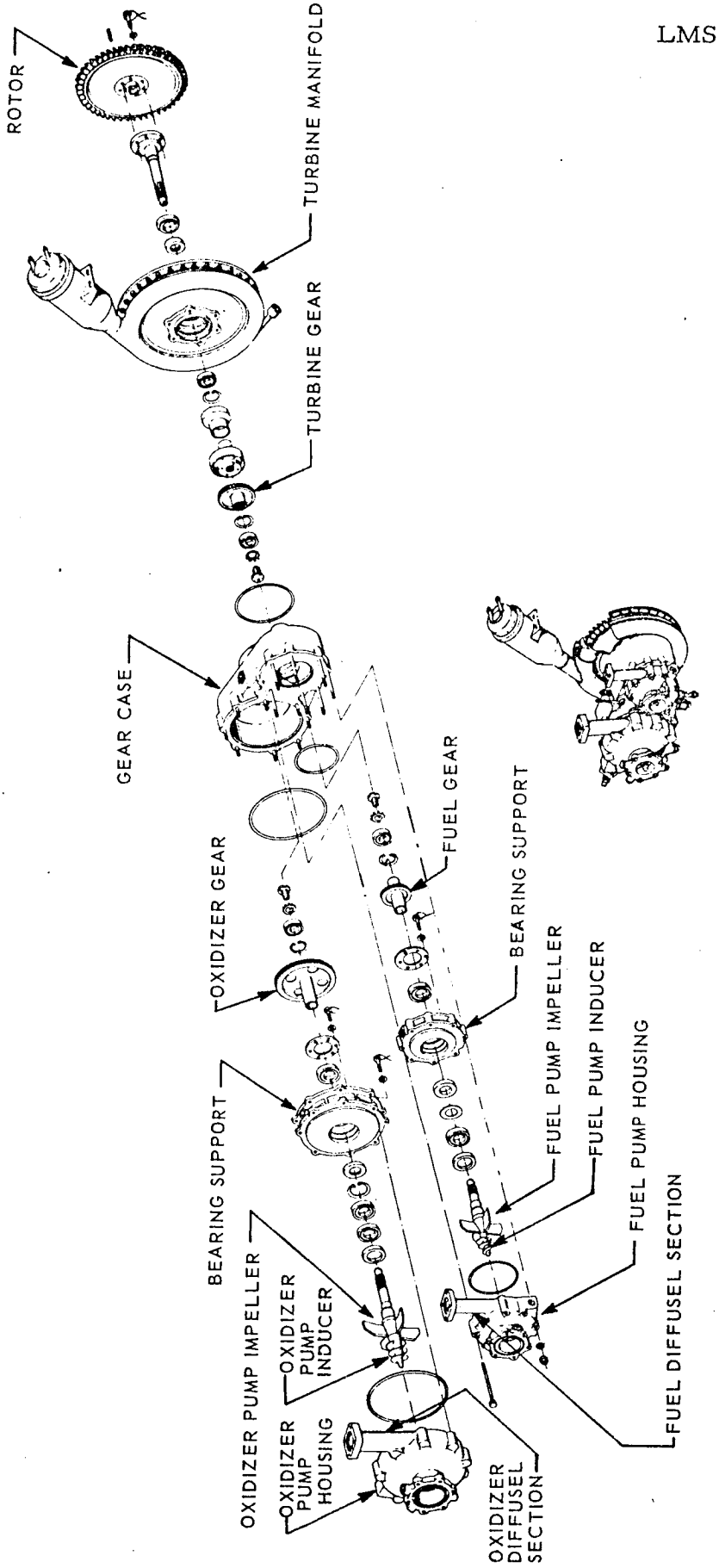


Fig. 2-19 Turbopump Assembly

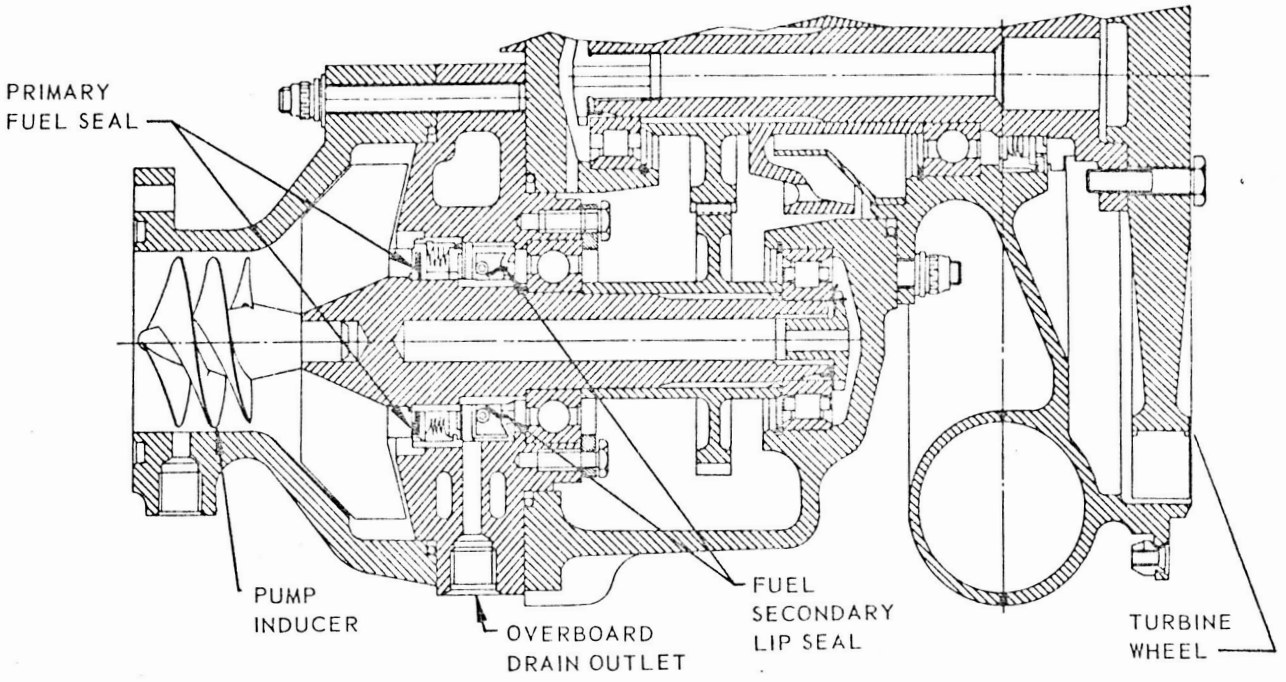


Fig. 2-20 Cross Section of Fuel Pump

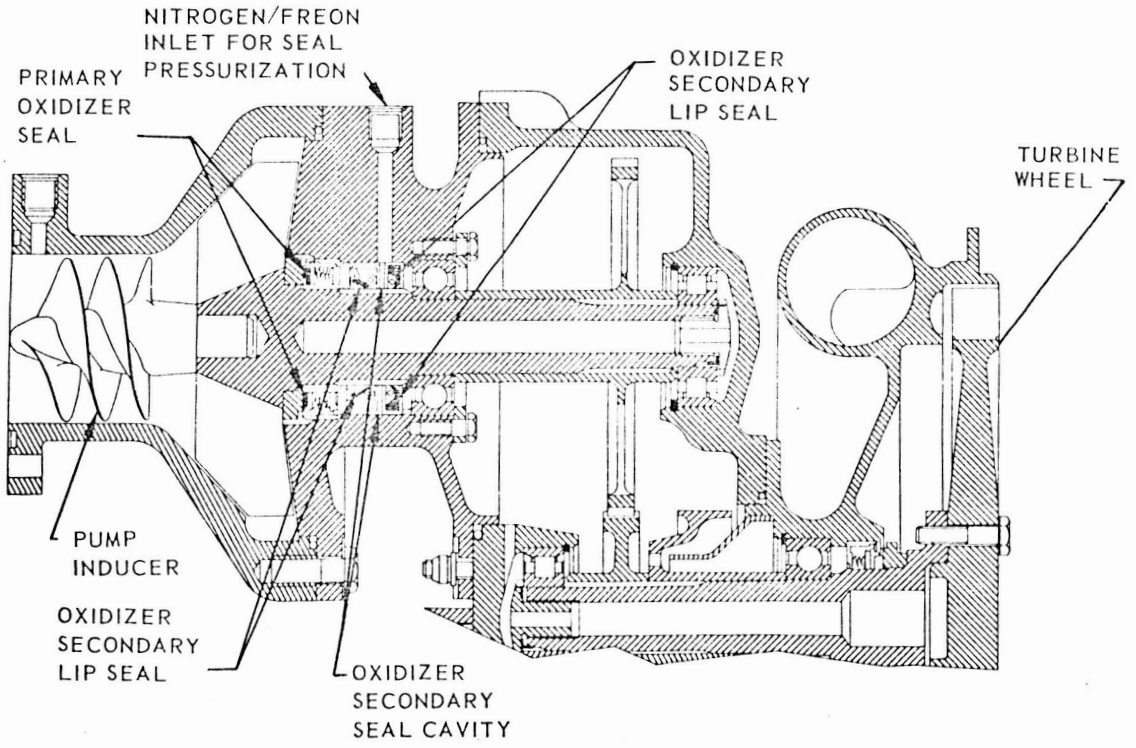


Fig. 2-21 Cross Section of Oxidizer Pump

2.2.7.2 Propellant Pumps. A description of the propellant pumps follows.

Design and Operation – The centrifugal propellant pumps (Fig. 2-19) incorporate circular castings and straight vane impellers with the flow terminating in venturi-type diffusers. These diffusers are designed to cavitate, thus giving the pumps the desirable characteristic of flow rate in direct proportion to speed, independent of head, within the operating range of the pumps. Although this design is not as efficient as pumps of conventional design, it eliminates the need for complex head suppression devices to control propellant ratio. Consequently, increased reliability and reduced weight are achieved with only a small reduction in pump efficiency.

Integral steel impellers and shafts (Fig. 2-19) are used in both fuel and oxidizer pumps. Aluminum inducers are pinned to the pump impellers to reduce the required net positive suction head for the pumps.

Turbopump Head Requirements – The minimum total inlet head required above vapor pressure at 60°F is 32.5 feet (11.15 psi) for the fuel pump and 11.5 feet (7.82 psi) for the oxidizer pump during start and operation. Figure 2-22 shows the vapor pressure variance with temperature for both propellants. At 60°F, minimum inlet pressure is 13.05 psia for fuel and 9.80 psia for oxidizer. The design speed for the fuel pump is 25,389 rpm and for the oxidizer pump is 14,410 rpm. The nominal total flow from the pumps is 15.27 lb/sec for fuel and 39.33 lb/sec for oxidizer.

2.2.7.3 Turbopump Assembly Lubrication System. The turbine gear box contains 150 cc of lubricating oil in accordance with Specification MIL-L-7808E. During the engine operation, bearing and gear lubrication is maintained by an oil mist formed by agitation of bulk oil by the rotating parts.

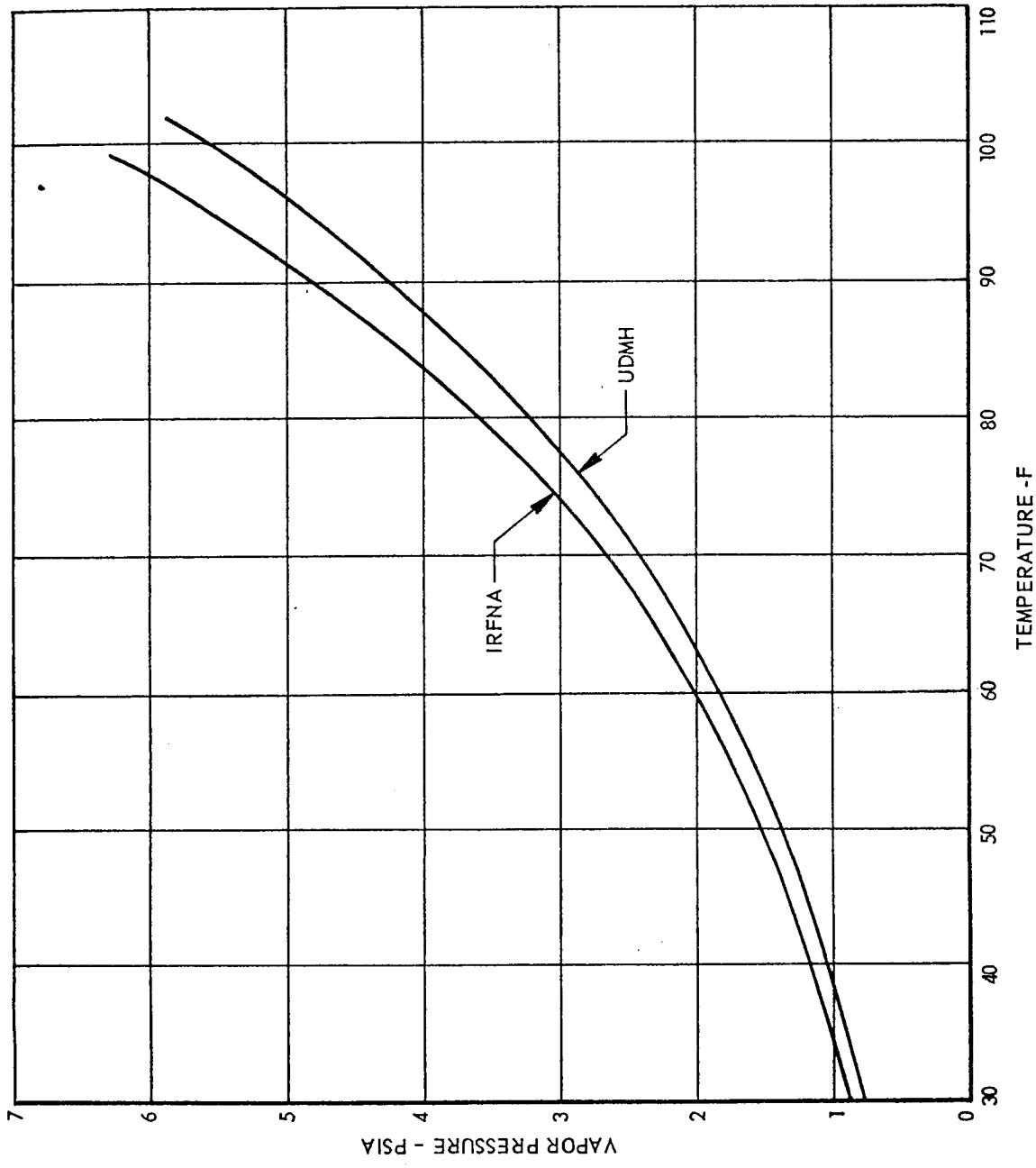


Fig. 2-22 Vapor Pressure Versus Temperature (UDMH and IRFNA)

2.2.7.4 Engine Gimbal Accessory Drive. A fuel tangential discharge port is provided on the fuel-pump housing to supply power to the positive displacement motor-pump hydraulic package. This hydraulic package, in turn, supplies hydraulic power for engine gimbaling. The fuel flow rate through the hydraulic power package is 6.75 ±0.5 gpm at a pump-discharge pressure of 1060 psia, but will vary directly proportionally to engine pump differential pressure. This flow requirement is satisfied with an orifice which is sized to have pressure-drop characteristics of 365 ±5 psi using UDMH with a specific gravity of 0.795.

Since the hydraulic system maintains almost constant actuation pressure and flow, there is negligible effect on the main engine system except for the consumption of a small amount of gas generator propellant to support the added load. The increase in gas generator flow rate for powering the auxiliary turbopump is computed at approximately 0.052 lb/sec.

2.2.8 Pilot-Operated Solenoid Valve and Fuel Valve

Pilot-Operated Solenoid Valve (POSV) Purpose and Operation – This valve (Fig. 2-23) is a normally open solenoid valve which is actuated by the engine-start signal, simultaneously with the gas generator solenoid valves. The valve is mounted integrally on and controls the opening of the fuel valve.

Fuel Valve Purpose and Operation – The fuel valve (Fig. 2-24) is a normally closed spring-loaded poppet-type valve operated by fuel-actuation pressure from the POSV, which admits fuel to the thrust chamber. This fuel valve contains a frangible disc to prevent UDMH from entering the thrust chamber and is ruptured by 525-psig fuel-pump pressure on the initial engine start. Maximum fuel valve opening pressure is 365 psi.

POSV And Fuel Valve Assembly Operation – Prior to engine start, the POSV poppet is in the normally closed position, and the POSV main poppet is in its

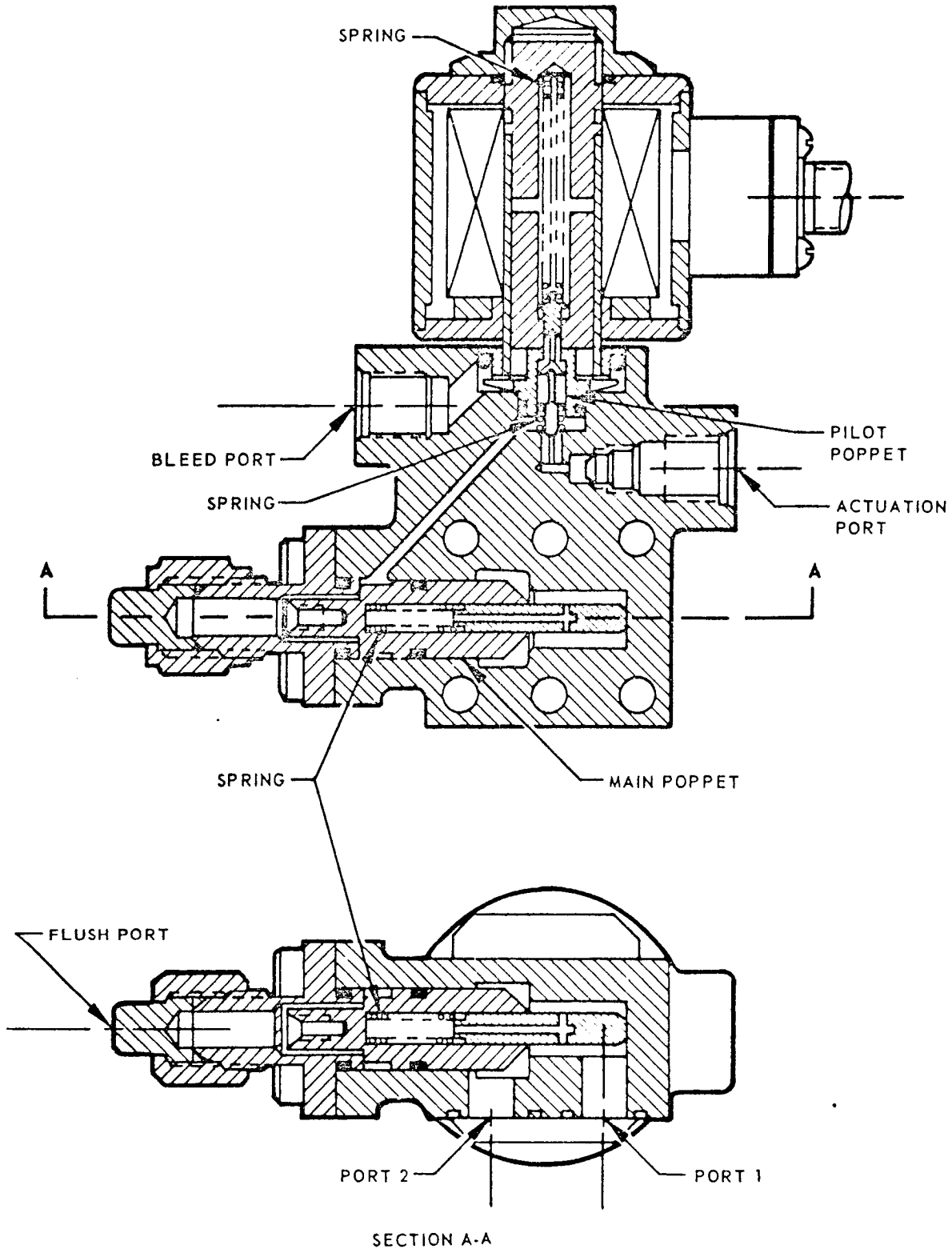


Fig. 2-23 Pilot-Operated Solenoid Valve

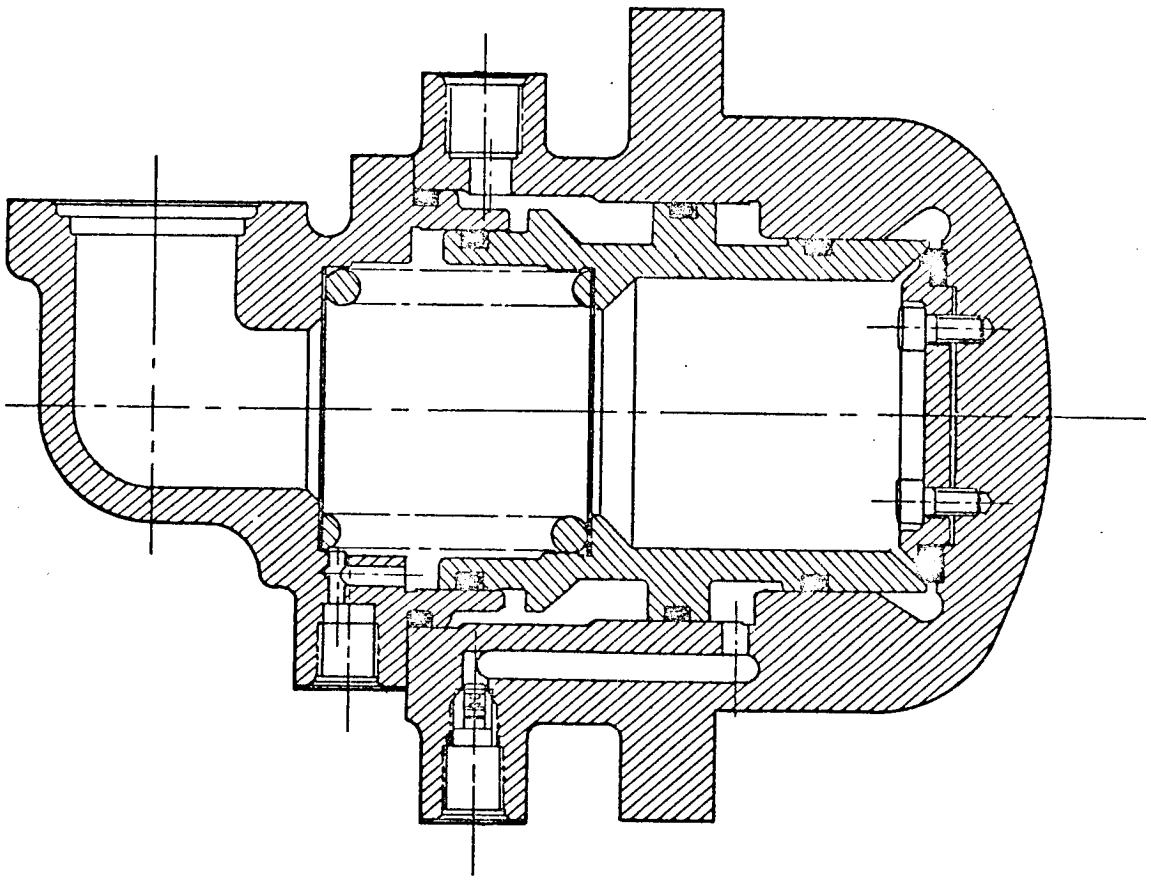


Fig. 2-24 Fuel Valve

normally open position (Fig. 2-25). Upon application of the start signal, the POSV solenoid coil is energized which opens the POSV poppet (simultaneously with the opening of the GG solenoid valves). As the fuel pump discharge rises to operating pressure, fuel-actuation pressure flows through port B of the fuel valve, through the actuation chamber, out through port A, and returns to the fuel pump inlet. Simultaneously, fuel enters the POSV through port D; since port C is blocked by the poppet, the fuel flows into the POSV main-poppet actuation cavity. Further fuel pressure buildup causes the POSV main poppet to close, thus shutting off flow through the main fuel-valve actuation cavity out port A, thereby allowing pressure to build up in the main fuel-valve actuation cavity. The fuel valve opens before this pressure reaches 350 to 375 psig, shuttling the main fuel-valve poppet and compressing the main actuating spring. On the initial start, the fuel valve opens and then the frangible disc ruptures – allowing flow into the thrust chamber fuel-manifold when discharge pressure rises to 525 psig. Upon all subsequent engine starts, fuel begins to enter the fuel manifold as soon as sufficient discharge pressure, 350 to 375 psig, is available to open the fuel valve.

The shutdown sequence is just the reverse of the start sequence. When electrical power is removed from the POSV, the POSV poppet closes under spring load. Actuation pressure entering port D to the POSV main poppet is bled off through port C and the POSV main poppet opens. This allows the fuel-valve actuation pressure entering port B to decay through Port A, and the fuel valve closes under spring load.

2.2.9 Oxidizer Valve

Purpose and Operation – The main oxidizer valve (Fig. 2-26) is a normally closed spring-loaded poppet-type valve, which admits oxidizer to the thrust chamber. Oxidizer pump-discharge pressure on the upstream valve face forces the valve open against spring pressure; opening pressure is

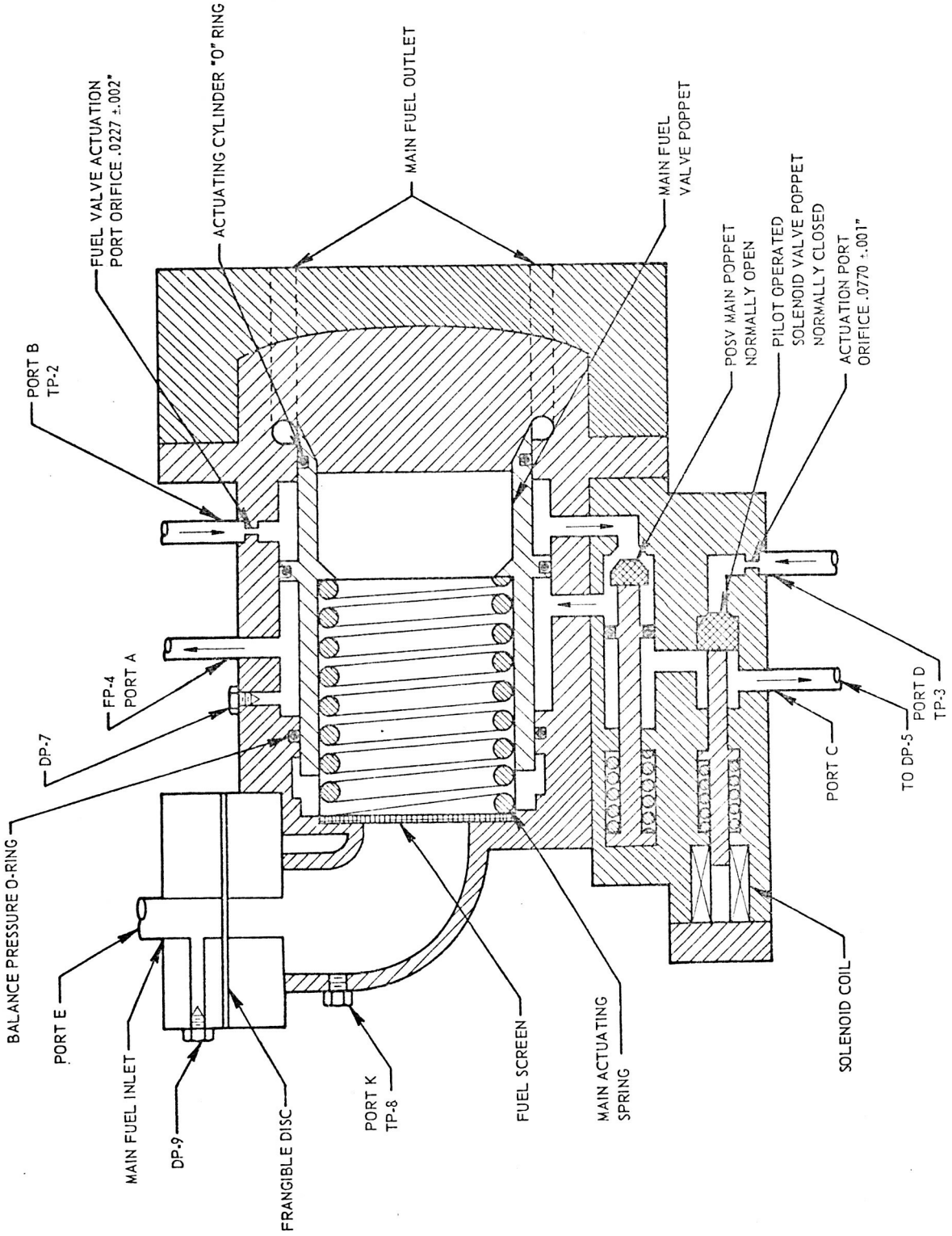


Fig. 2-25 Pilot-Operated Solenoid Valve and Fuel Valve

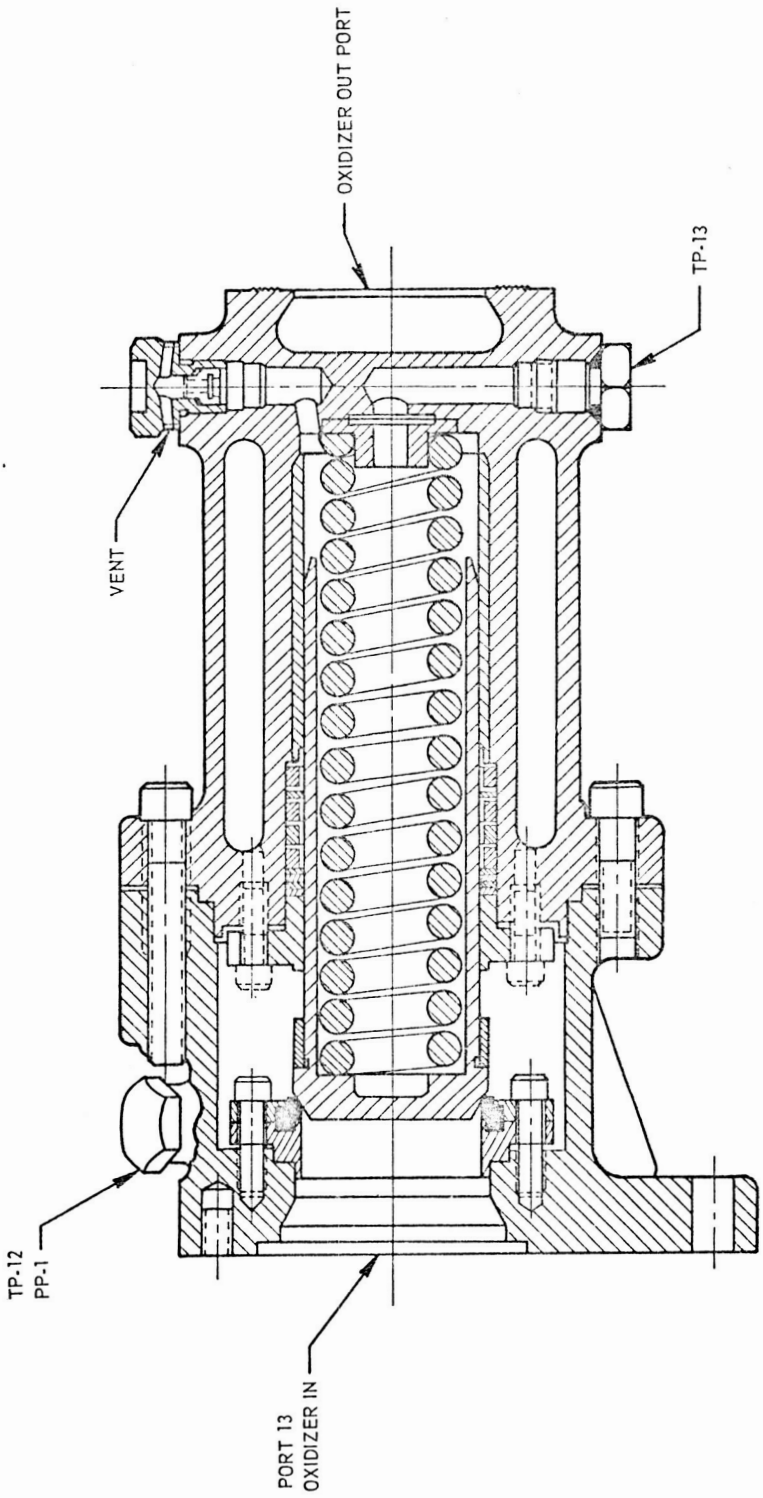


Fig. 2-26 Oxidizer Valve

225 to 300 psig. A frangible disc in the downstream side of the valve guarantees zero leakage prior to first engine start. The frangible disc will rupture at a pressure of approximately 180 psi. On shutdown, decay of the upstream turbopump discharge pressure allows the valve to close under spring pressure.

Effects of Valve Spring Strength on Oxidizer Postflow – Studies were made to determine the effect of oxidizer valve closing times on oxidizer postflow. For a decrease from the nominal value of 2.2 sec (based on 130-psi cracking pressure) to 0.5 sec, postflow would decrease from 20 lb to about 7 lb. Decreasing the valve stroke afforded insignificant change over the range studied; however, increasing the spring force showed significant effects on postflow. Minor modifications to the valve increased the cracking pressure range from 225 to 300 psig. Preliminary calculations indicate the total shutdown impulse to be 2300 (+600, -900) lb-sec.

2.2.10 Engine Thrust-Chamber Assembly

Design and Operation – USAF XLR-81-BA-13 engine thrust chamber (Fig. 2-27) is an 80-percent bell-shape with an expansion ratio of 45:1. This thrust-chamber assembly performs satisfactorily throughout a gimbaling orientation of plus 5 deg in a square pattern from the associated planes, with a gimbal acceleration of 30 rads/sec/sec. The nozzle is regeneratively cooled up to the point at which the area ratio is 13.3:1. The remaining portion of the nozzle (13.3:1 to 45:1) is a radiation-cooled titanium extension.

Thrust Chamber Construction – The thrust chamber is fabricated of 6061 aluminum alloy, in three sections: the combustion chamber, nozzle throat, and divergent nozzle section. Each section is machined with integral cooling

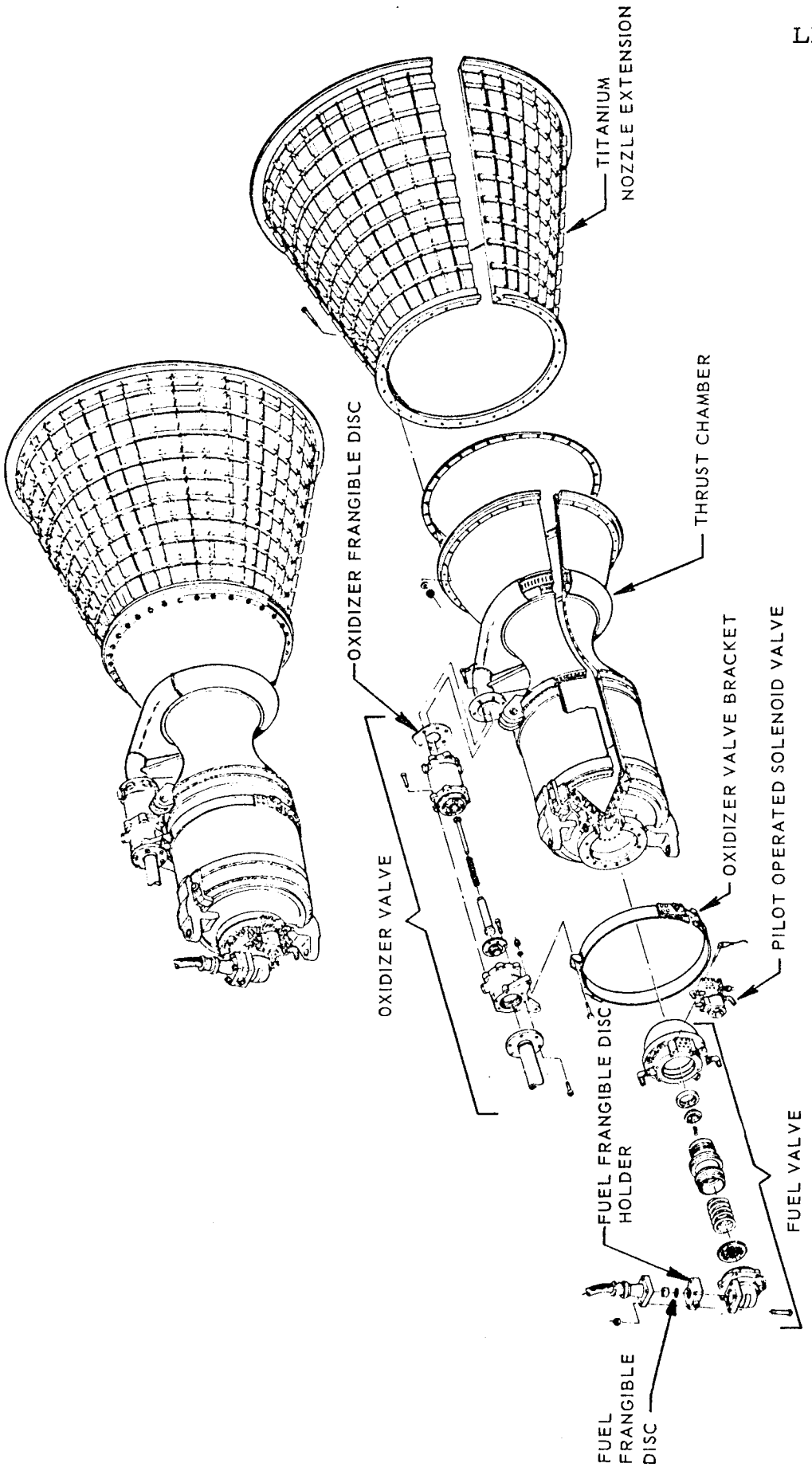


Fig. 2-27 Thrust Chamber Assembly

passages drilled through the walls of the combustion chamber and nozzle sections. These passages are drilled in the combustion chamber in an axial direction around the periphery. The holes in the throat section are skewed to form a hyperboloid, and holes in the divergent nozzle are drilled to conform to the contoured configuration. After matching, these sections are heli-arc-welded into an integral assembly. The absence of common welds separating the propellant passages, through which leakage might occur, is a special feature of this integral design, which provides a rugged structure; eliminates the need for external joints; and furnishes a smooth interior for reducing turbulence, heat-transfer rates, and performance losses. The inside surface of the combustion chamber is coated with aluminum oxide. The radiation-cooled titanium nozzle extension is bolted to the regeneratively cooled section of the nozzle at the point of 13.3:1 area ratio. The titanium nozzle extension is coated on the inside with an aluminum-oxide refractory ("Rokide") and finished on the outside with a high-emissivity surface. This extension is reinforced with molybdenum bands and stringers as shown in Fig. 2-28. Nozzle extension failure due to the thermal shock incurred at each firing is the limiting factor controlling the number of restarts of the main engine. The nozzle throat area is approximately 17 sq in.

Thrust-Chamber Flow Passages – The oxidizer enters the thrust-chamber cooling passages through a manifold ring located slightly aft of the nozzle throat. Flow is divided and equalized in this ring by baffles. All of the flow traverses to the exit end and then flows forward through the full length of the nozzle and chamber and into the injector manifolding.

The coolant oxidizer passes through 72 holes of 0.172-in. diameter to the exit-end manifold of the divergent section and returns through a separate set of 72 holes (also 0.172-in. in diameter) to the plenum chamber (manifold) adjacent to the inlet manifold. From the plenum chamber, the oxidizer passes through 75 holes of 0.116-in. diameter to the transfer manifold for

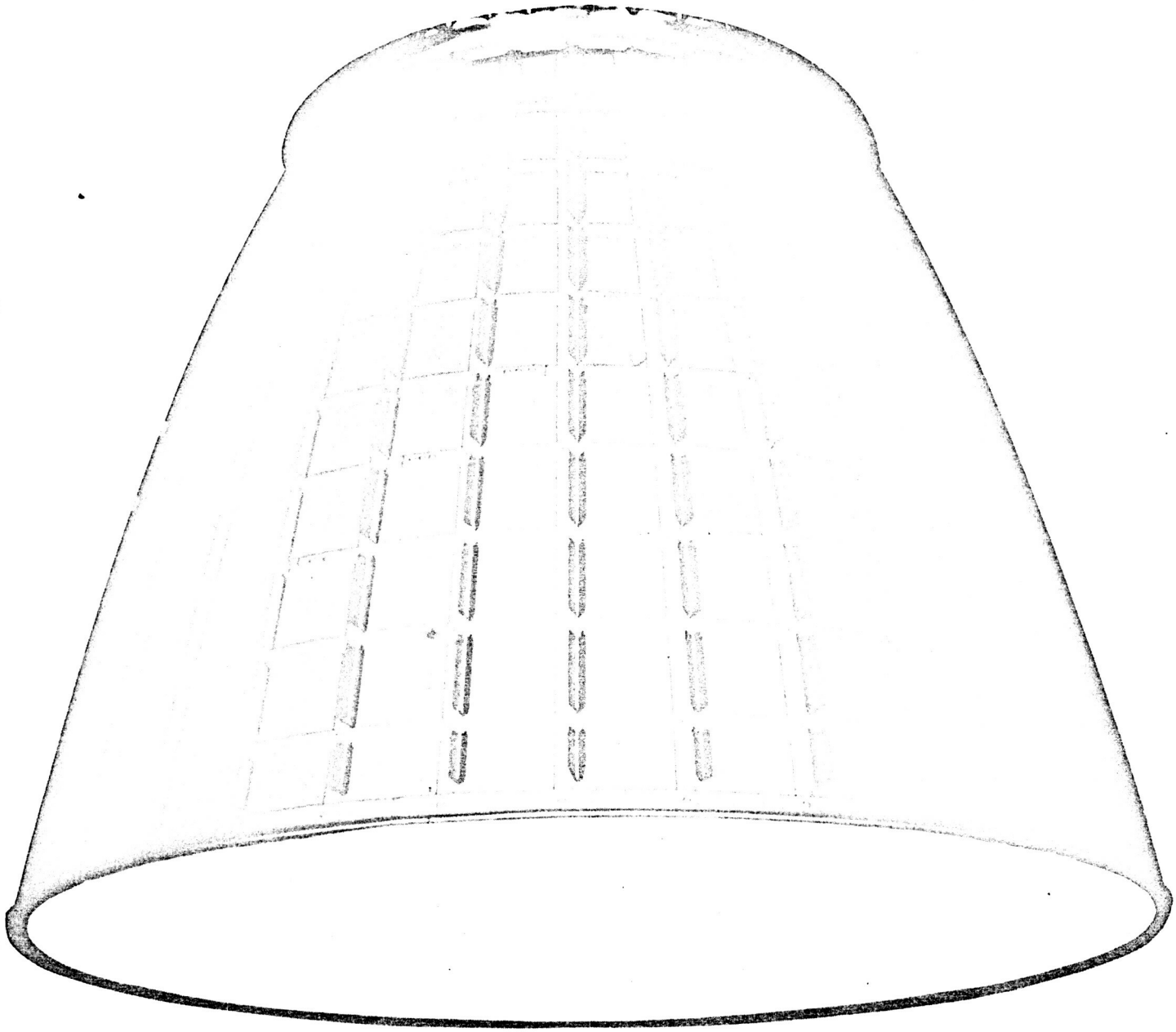


Fig. 2-28 Thrust Chamber Nozzle Extension

cooling the throat section. Forward of the transfer manifold, 185 coolant holes of 0.125-in. diameter lead to the injector manifolding.

Thrust-Chamber Injector – The injector (Figs. 2-29 and 2-30) is a triplet type with two fuel jets impinging on each oxidizer jet and a ring of fuel doublets around the perimeter to provide film cooling. The injector body is machined from an aluminum-alloy forging and is heli-arc-welded to the thrust-chamber assembly. It has a cavity in which the fuel valve fits; this allows the fuel valve poppet to be located close to the fuel injector orifices. Free volumes are thus held small, and when the fuel valve opens, fuel injection starts with minimum delay and with minimum "hammer" as the small void fills with propellant. A similar arrangement is not needed for the oxidizer, since oxidizer flow is accelerated gradually.

2.2.11 Engine Mount and Gimbal Assembly

Design and Operation – The engine mount consists of a nickel-plated tubular-steel frame for attachment at four points. The thrust-chamber assembly is attached to the engine mount on a mutually perpendicular, two-axis gimbal system providing vehicle pitch and yaw control. The gimbal mount system provides ±5 deg of deflection in a square pattern; however, the hydraulic actuators limit movement to ±2 deg, 30 min in a square pattern.

A ring containing four equally spaced bearings encompasses the thrust chamber forward of the injector. Pivots for these bearings are provided by close tolerance 0.7500-in.-diameter pins, two of which are positioned in lugs welded to the forward face of the injector. The pins consist of a bolt-head and thread at opposite ends and are maintained in proper position with lock nuts. Rotation around these two pins provides deflection in one plane. In a similar manner, two additional pins attached to the engine-mount structure permit deflection in the other plane. Thrust is thus transmitted through each

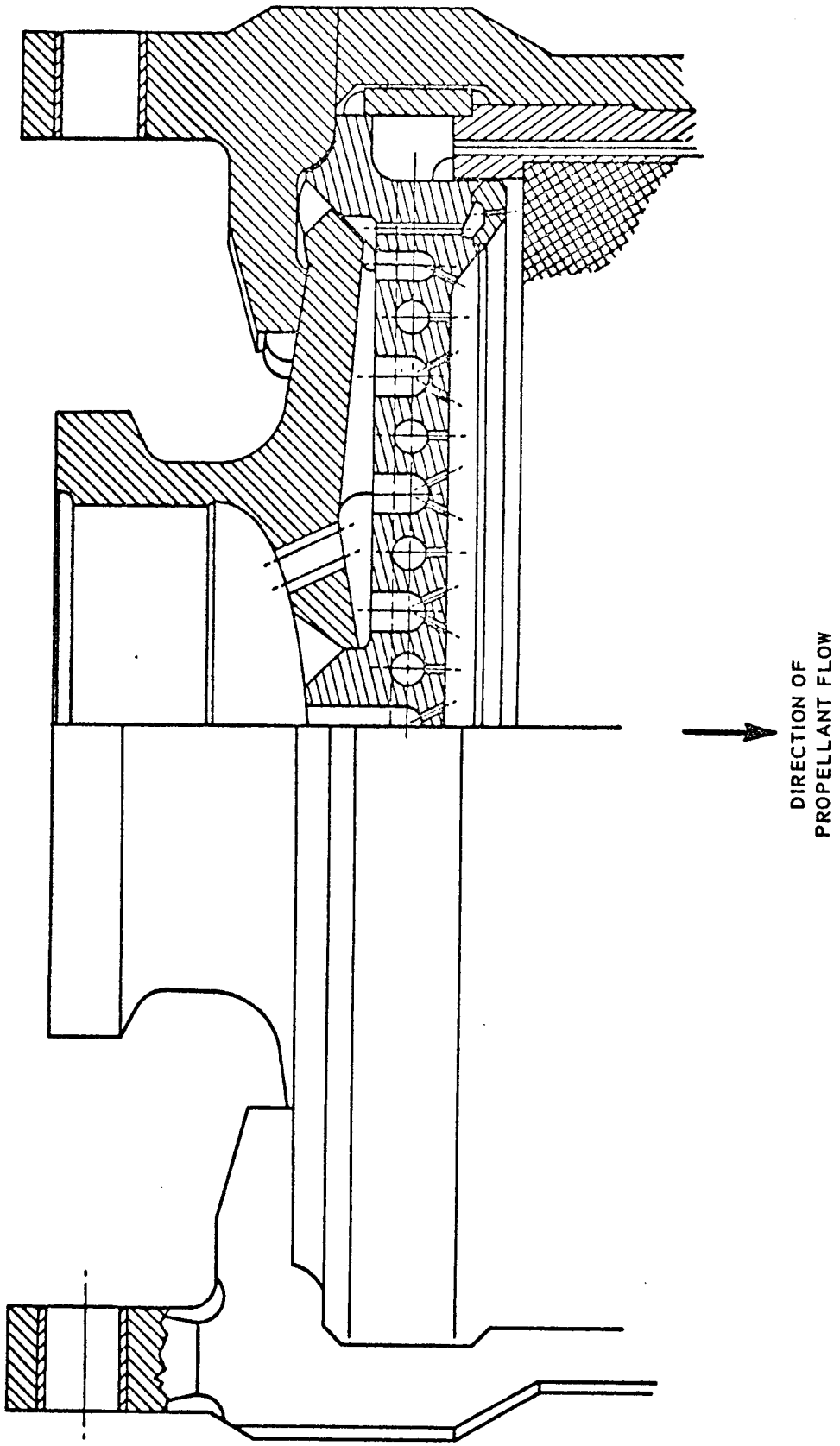


Fig. 2-29 Cross Section of Engine Injector

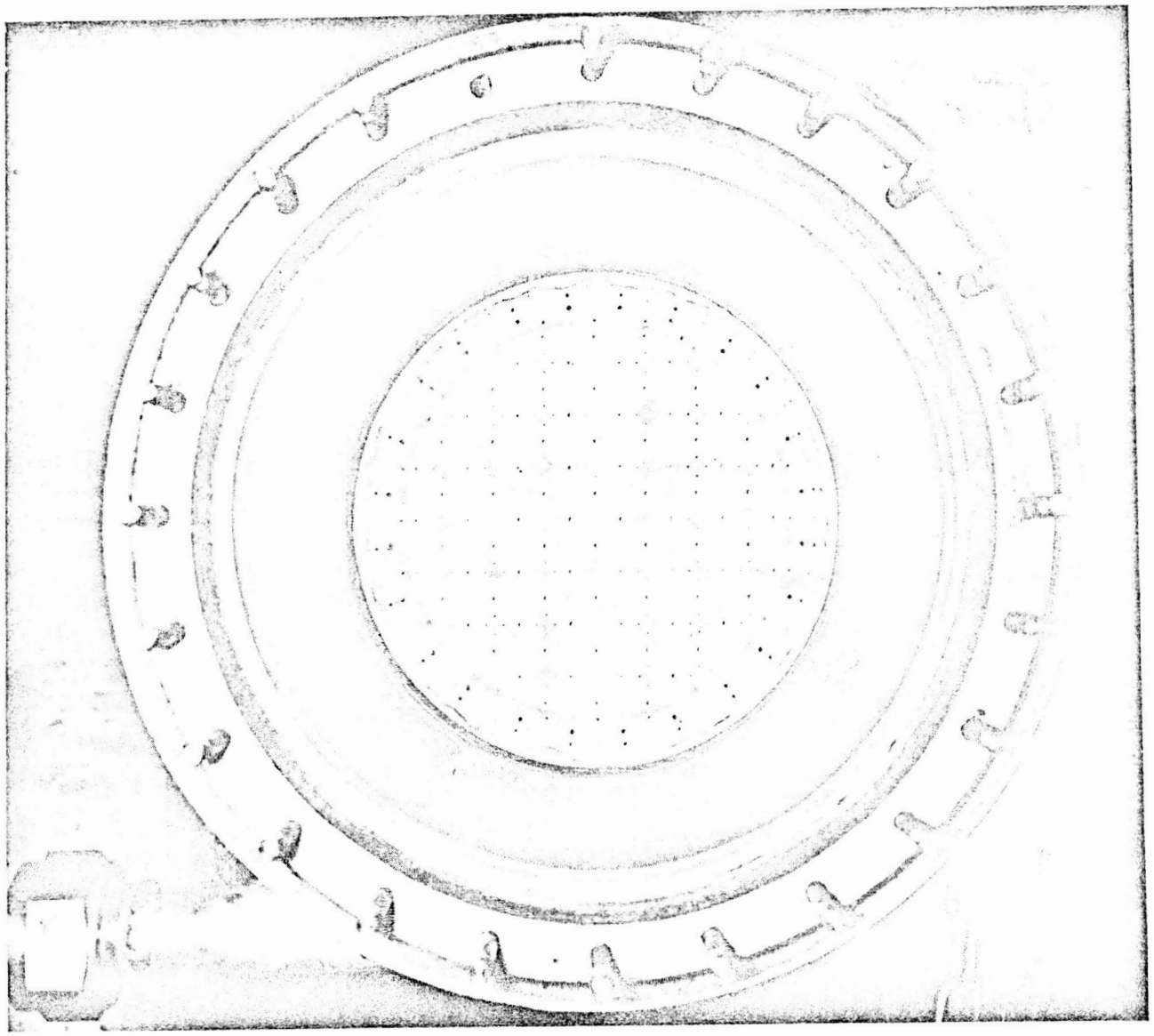


Fig. 2-30 View Showing Engine Injector

pair of thrust-chamber and engine-mount bearings. Flexible tubing for the propellants connects the pumps and thrust-chamber assembly. Figure 2-31 shows gimbal attachment to the engine mount.

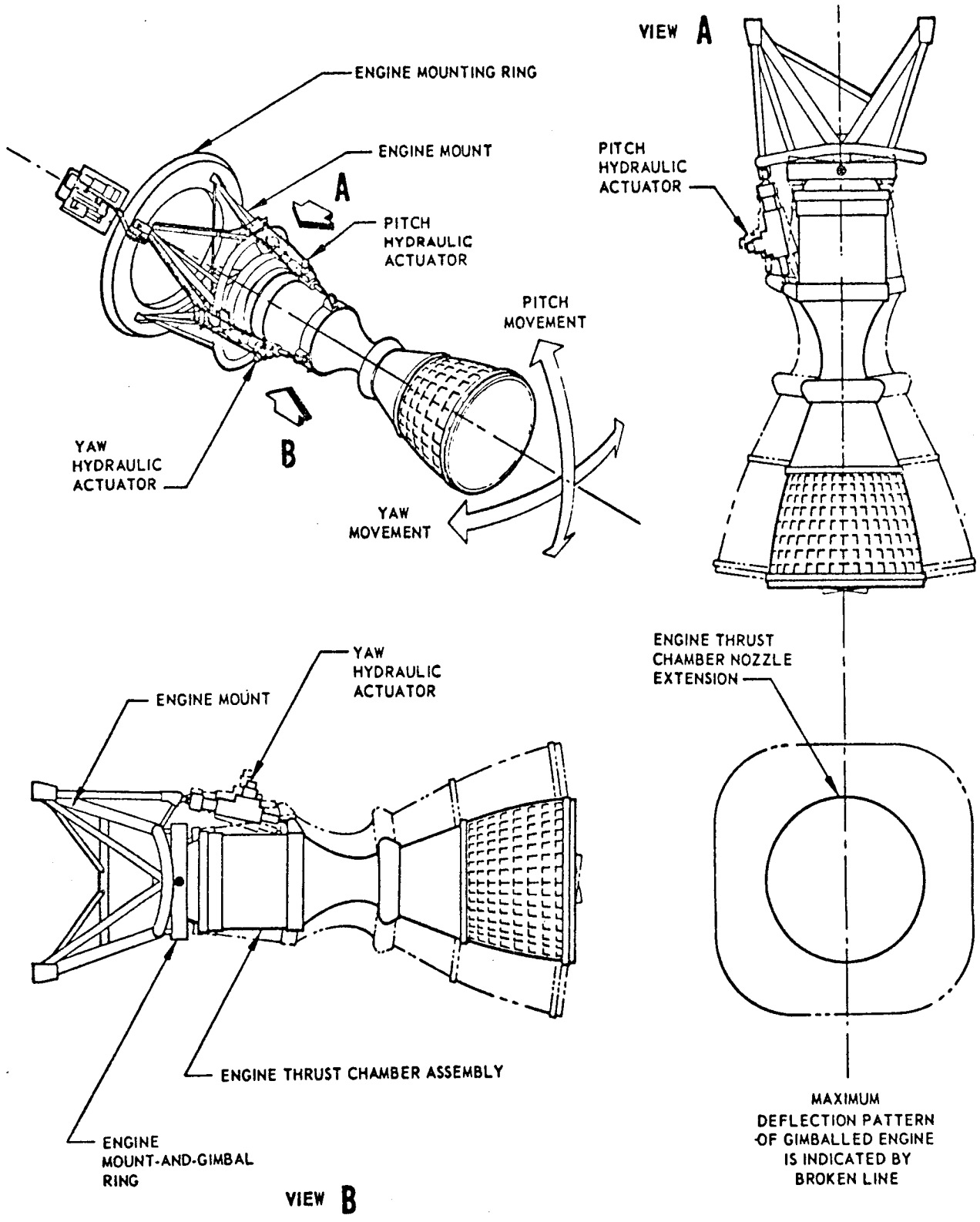


Fig. 2-31 Engine Gimbal Attachment and Movement

2.3 PROPELLANT FEED AND LOAD SYSTEM

The fuel and oxidizer feed and load systems consists of the following major components:

- Propellant tank assembly
- Propellant containment system
- Propellant isolation valves (PIV's)
- Propellant fill and drain quick disconnects
- Propellant fill bellows
- Propellant pump-inlet feed bellows

The feed and load system provides controlled means (1) of loading propellants into the vehicle tanks prior to launch and (2) of supplying propellants to the engine during flight.

2.3.1 Propellant Tank Assembly

Purpose – The propellant tank assembly (Fig. 2-32) stores liquid propellants in sufficient quantities to allow accomplishment of total Gemini-ATV mission objectives. Tankage consists of a dual-chamber assembly, having an integrally mounted passive fuel and oxidizer containment system or "sumps." (See Section 2.3.2.) Fuel and oxidizer chambers are separated by a common bulkhead, making necessary the maintenance of a nominal 2-psid pressure differential of fuel over oxidizer to preclude bulkhead reversal.

Operating Pressures and Nominal Volumes – The nominal operating fuel and oxidizer pressure is 55 psig each, with a proof pressure of 61 psig. Minimum net capacities are 75.9 ft^3 for the fuel tank and 98.4 ft^3 for the oxidizer tank. Maximum weight of the propellant tank assembly alone is 315 lb.

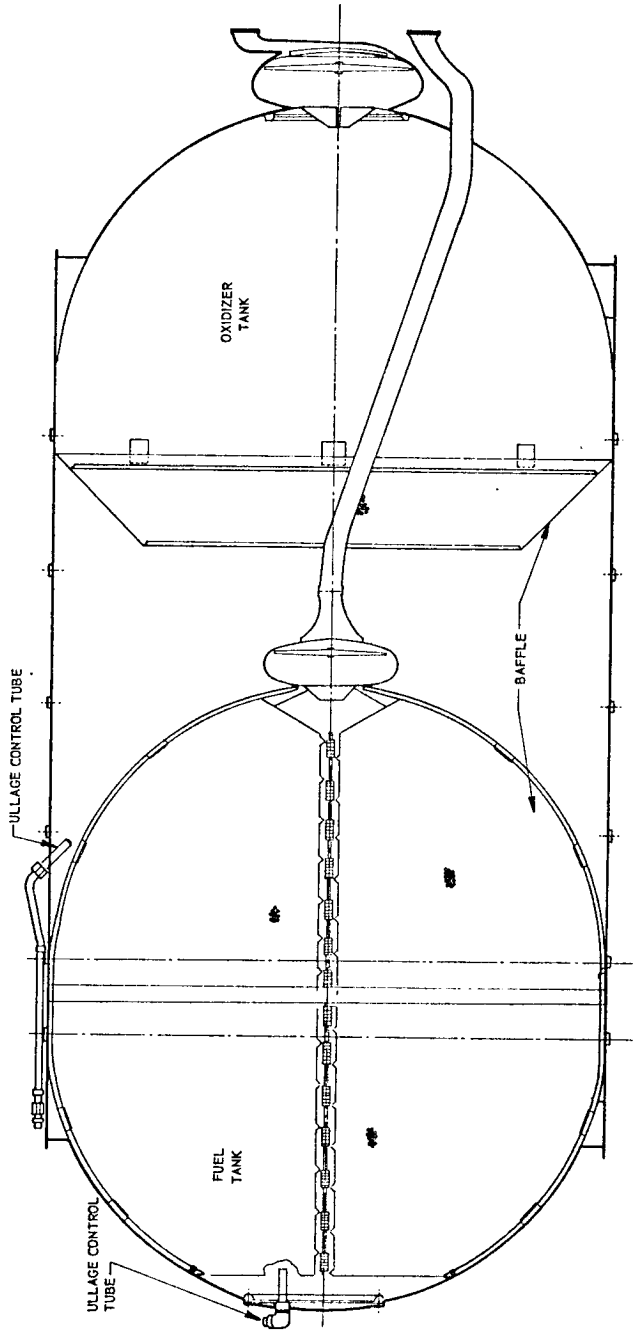


Fig. 2-32 Propellant Tank Assembly

2.3.2 Propellant Containment System

Purpose – The propellant containment system, commonly referred to as fuel or oxidizer "sumps," is shown in its relationship to the propellant tanks in Fig. 2-32. This containment system provides improved propellant scavenging and passive propellant orientation.

Operation – The concept involves scavenging propellants at depletion and containment of sufficient propellants to start and operate the main engine until the bulk propellants in the tanks become oriented over the pump inlets. A desire to minimize tank-trapped propellant residuals (non-impulse propellants) and to simplify the systems led to the design.

Figure 2-33 shows the fuel sump configuration and Fig. 2-34 depicts the oxidizer sump design. The fuel sump has a nominal capacity of 0.25 ft^3 while the oxidizer sump has a capacity of 0.56 ft^3. These sumps provide approximately two sec of additional burn time by adding their volumes to those of the tanks, and by improved scavenging – therefore reducing the quantity of non-impulse propellants. Design pressures are identical to those for the propellant tanks.

The containment feature of the sumps is not considered mandatory to ATV mission requirements, because the secondary propulsion system is utilized for propellant orientation (discussed in Section 4.1 of this report). However, since minimum program peculiar changes were desired to the Standard Agena-D Vehicle, the containment system was left intact. This containment system offers no detrimental effects to the ATV nor does it influence mission requirements or objectives.

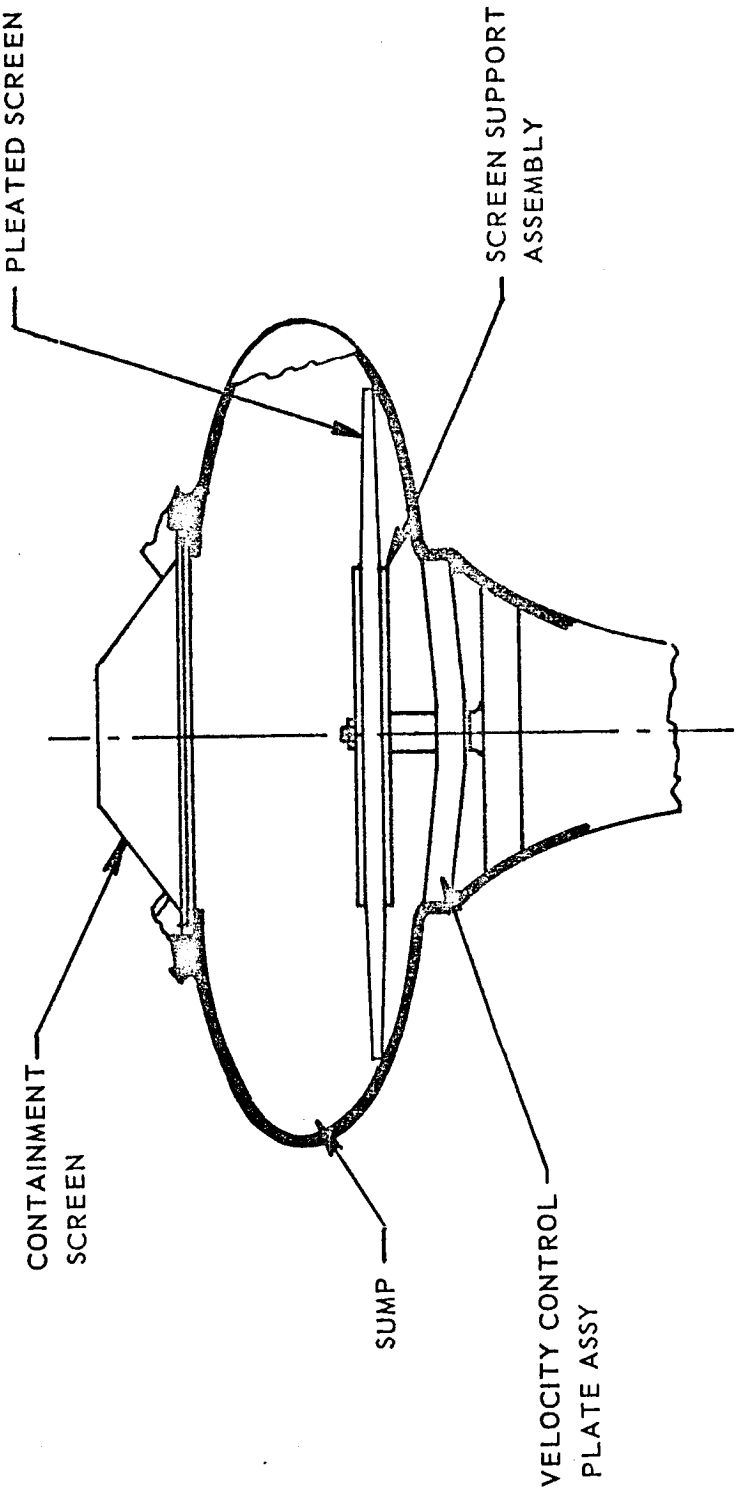


Fig. 2-33 Fuel Sump (Propellant Containment System)

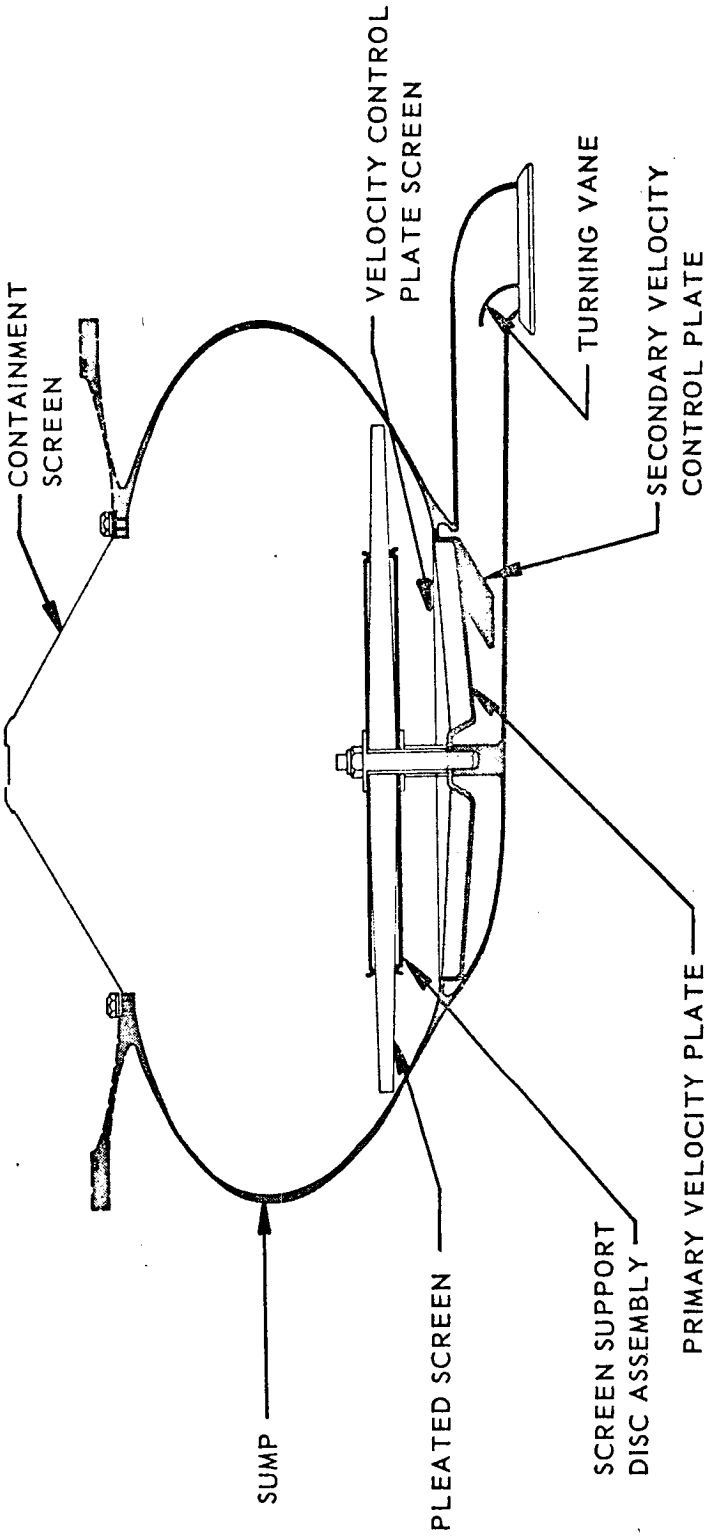


Fig. 2-34 Oxidizer Sump (Propellant Containment System)

2.3.3 Propellant Isolation Valves

Operation – The oxidizer and fuel propellant isolation valves (PIV's) are motor-operated blade valves located between the propellant tank outlets and the propellant pump inlets (Fig. 2-2). In the open position, these valves allow filling, draining, and flushing of the tanks through an integral fill port located on the engine side of the blade. When closed, the PIV's isolate the turbopump inlets and fill lines from the propellant tanks and vent engine-entrapped residual propellants through an integral vent port which opens only after the valve is fully closed. The vented propellants are routed through lines running parallel to the turbine-exhaust duct and terminating in nullifiers, thereby precluding the possibility of an increase in vehicle velocity after shutdown of the main engine and subsequent closing of the fuel and oxidizer PIV's.

On application of 28 vdc, the valves actuate from the open to closed position, or from closed to open, by means of an integral electric motor which is de-energized at the fully open and the fully closed positions. The valves incorporate an external visual-position indicator and an electrical connector to provide for remote electrical indication and operation. On command, these valves cycle repeatedly from open to closed or closed to open. The present sequence of events indicates that the ATV will be launched with the PIV's open. The propellant isolation valves will close approximately six sec after each engine shutdown and upon command reopen four sec prior to each engine start.

These valves were designed for use on Gemini vehicles but will be used for all dual- or multiple-burn missions on future Agena vehicles.

The integrally mounted electric motor (Fig. 2-35) drives a planetary gear train which is connected to a shaft. Rotary motion of the shaft is converted

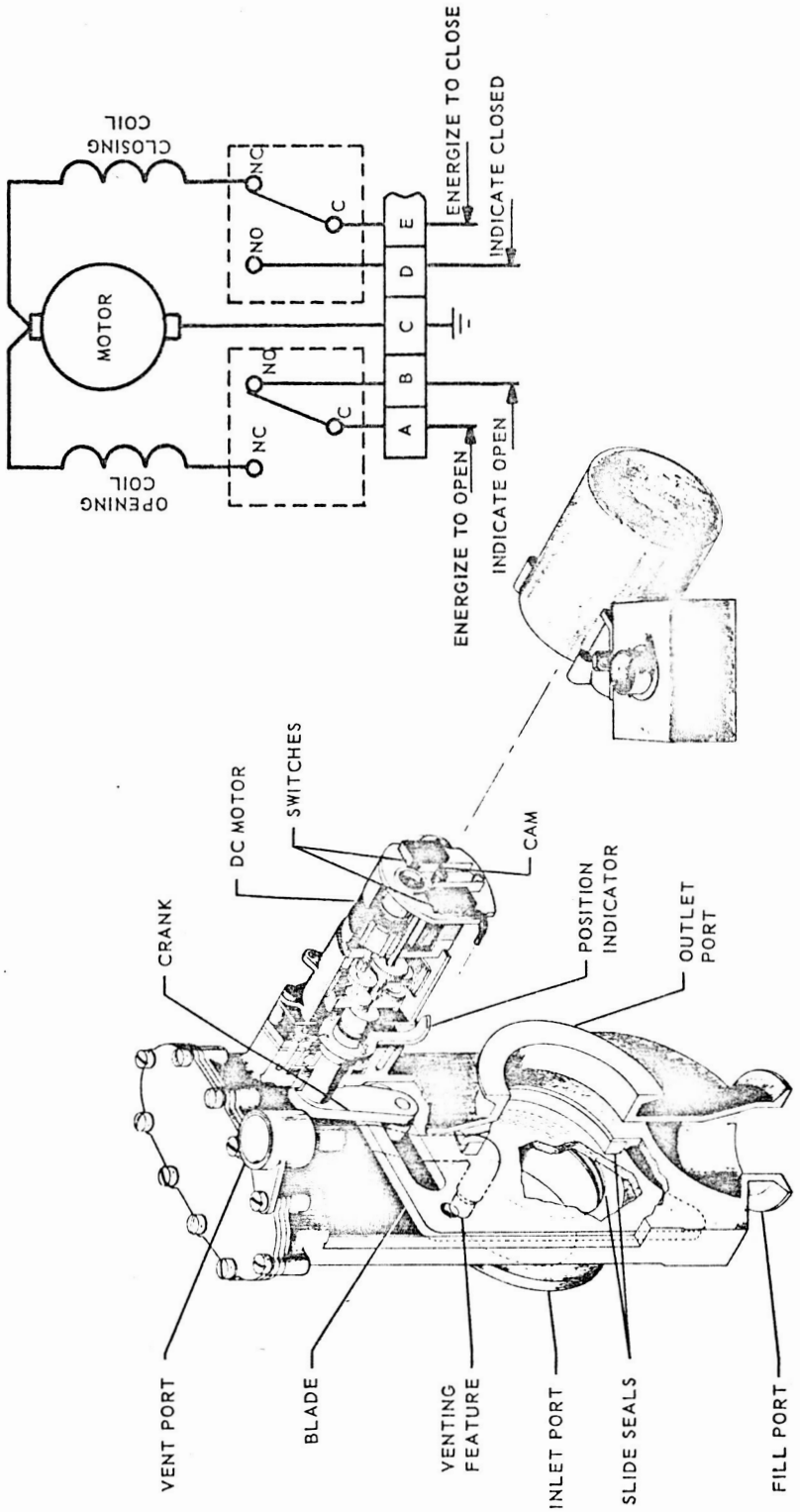


Fig. 2-35 Propellant Isolation Valve

to linear motion by means of the crank fixed on the shaft at one end, with the engine end sliding in the horizontal slot in the blade. The crank moves through an arc while retained in the blade slot. This action forces the blade to move in a vertical direction, and thus slide against the seals retained in the valve body; the blade's sliding action opens and closes the flow area. The venting mechanism opens only after the valve blade has fully closed off the flow area, and closes before the valve blade opens to allow propellant flow.

While on the launch pad, the valves are in the open or "fill" position which is maintained at liftoff and in flight prior to first burn. A redundant momentary open signal is sent to the valves in preparation for engine start in order to assure that the valves are open and ready for engine operation.

The first PIV units were developed as a dual-source item by Whittaker Controls and Guidance (WCG) Division of Telecomputing Corporation, and by Reaction Motors Division (RMD) of Thiokol Chemical Corporation. Both units, identified by the LMSC P/N 1463144-1, were functionally and physically interchangeable with the exception that each vendor had a basically different venting mechanism, which resulted in a different external location for the vent port. Therefore, the RMD valve required longer vent tubes than the WCG valve. This difference led to designating the RMD unit as P/N 1463144-3.

Problems with IRFNA salts building up in the WCG-1 valve during long-storage periods brought about a redesign of the gear train, which resulted in the lowering of the final drive ratio, increasing the torque on the blade, and increasing the valve opening time from two to four sec. This latter valve is the P/N 1463144-5. The WCG-1 valve will no longer be used.

In summary, there are two propellant isolation valves which may be used on the Gemini-ATV vehicles. These are the RMD-1463144-3 or the WCG-1463144-5.

Table 2-3 presents the operating characteristics of the propellant isolation valves.

Table 2-3

PROPELLANT ISOLATION VALVE OPERATING CHARACTERISTICS

1.	Flow Characteristics (with IRFNA at 60°F):	
	• Pressure drop:	
	Inlet to outlet	200 gpm - 0.50 psi
	Fill-port to inlet-port	60 gpm - 5.0 psi
	• Leakage through blade valve pressurized 0 to 60 psig	0.10 cch
	• Vent port flow at 50 psia	1 gpm
2.	Actuation time	RMD-3 - 2 sec max.
	(Extreme to extreme)	WCG-5 - 4 sec max.
3.	Actuation voltage	22 to 28 vdc unregulated
4.	Service life	1020 cycles
5.	Structural:	
	Proof pressure	150 psig
	Burst pressure	200 psig

2.3.4 Propellant Fill and Drain Quick Disconnects

Purpose — The fuel and oxidizer fill and drain quick disconnects (QD's) (Fig. 2-36), which are identical in design, provide access to the propellant tanks for loading or draining propellants under controlled conditions. These QD's are connected manually and disconnected remotely.

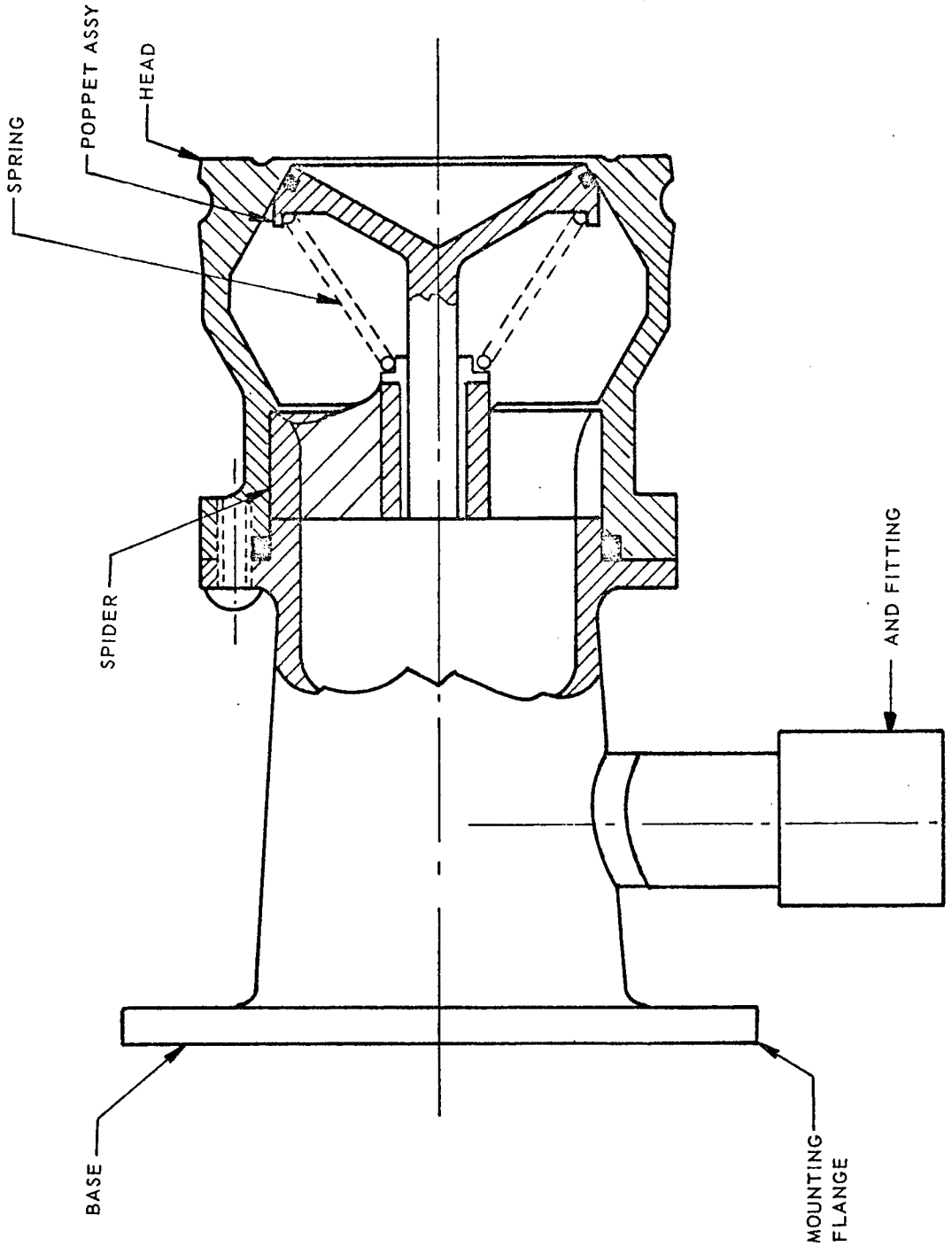


Fig. 2-36 Airborne Portion of Propellant Tank Fill and Drain Disconnect

Operation – The disconnect assemblies are of double-poppet design, with one poppet in the airborne and the other in the AGE portion of the assembly. As the halves are mated, a seal is established between them. The seating ends of both poppets contact each other and stay in contact as the poppets are forced open by the engaging action of the disconnect assembly. Locking is accomplished by a ball, detent, and sleeve-type ball-retainer arrangement. The disconnect is accomplished by a ground-controlled piston actuator which, when pressurized with gas, moves the sleeve away from the ball, thus allowing it to slip out of its detent (which locked the two halves together). Internal pressure then acts to force the two halves apart. The two poppets move together in the opposite or closing direction, and because essentially no free volume surrounds the poppets, virtually no fluid is present at the mating interface, thus spillage is minimized as the disconnects separate. If for any reason the pneumatic actuators fail to separate as the vehicle lifts off, a redundant lanyard action for separation is accomplished by a pull on external lines.

The airborne portion of the disconnect assembly is shown in Fig. 2-36. An auxiliary port on the side is for an optional "Propellant Dump Kit," which is not used on Gemini-ATV vehicles and is capped.

2.3.5 Propellant Fill Bellows

The fill bellows, mounted integrally to the disconnect assembly (Fig. 2-2), accommodates slight misalignments and contractions of the attached components. Within each fill bellows, a conical strainer is positioned to prevent the passage of foreign particles larger than 0.030 in. in diameter.

2.3.6 Propellant Feed Bellows

The feed bellows (Fig. 2-2), mounted between the propellant isolation valves and the turbopump inlets, serves the following functions:

1. Accommodates minor misalignments of the attached subassemblies
2. Isolates engine vibration
3. Absorbs thermal expansion and contraction of the attached subassemblies
4. Absorbs forward movement of the engine during firing
5. Absorbs expansion and deflection of propellant tanks and sumps when filled

Optional Bellows – Two sets of bellows for fuel and for oxidizer are available. The standard set accommodates misalignment from 0 to 0.140 in.; the alternate set accommodates misalignment from 0.140 to 0.280 in. and may be required due to a buildup of allowable manufacturing tolerances. The choice of bellows varies from vehicle to vehicle.

2.4 GAS PRESSURIZATION SYSTEM ASSEMBLY

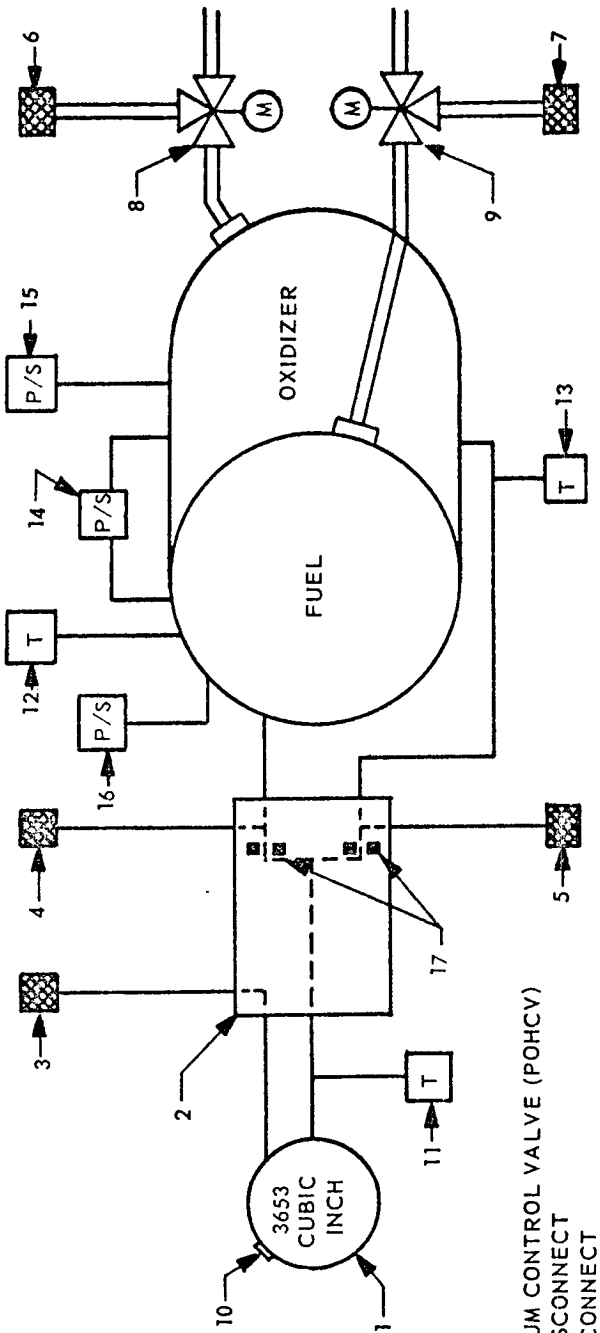
Description – The Gemini-ATV pressurization system (Fig. 2-37) supplies and maintains the desired pressures in the propellant tanks throughout flight. This pressurization system consists of the following:

(1) Helium-fill quick disconnect
(2) Propellant-vent quick disconnect
(3) Helium sphere (3653 $in.^3$ capacity)
(4) Pyro-operated helium-control valve
(5) Oxidizer-tank pressure switch
(6) Fuel-tank pressure switch
(7) Propellant tank ΔP pressure switch

Operation – The operation of the gas pressurization system consists of opening the helium control valve (first actuation) at some discrete time (nominally 1.5 sec) after the start signal is sent and closing the oxidizer portion of the valve (second actuation) approximately 318 sec later. Control of fuel and oxidizer propellant-tank pressures with time is altered (1) by two 0.027-in.-diameter orifices (built into the valve assembly) which control the discharge from the gas-storage sphere to the propellant tanks and (2) by control of the propellant ullage volumes and pre-pressurization of the propellant tanks prior to launch.

2.4.1 Helium-Fill and Propellant-Vent Quick Disconnects

Helium Fill Quick Disconnect – The airborne half of the helium-fill quick disconnect (QD) assembly, when coupled with the ground half, provides access to the vehicle-propellant pressurization-control system for loading and unloading of high-pressure gas. After separation of the halves and prior to first actuation of the helium control valve, the self-sealing airborne half prevents the loss of pressurization gas.



1. HELIUM SPHERE
2. PYRO OPERATED HELIUM CONTROL VALVE (POHCV)
3. HELIUM FILL QUICK-DISCONNECT
4. FUEL VENT QUICK-DISCONNECT
5. OXIDIZER VENT QUICK-DISCONNECT
6. OXIDIZER FILL AND DRAIN QUICK-DISCONNECT
7. FUEL FILL AND DRAIN QUICK-DISCONNECT
8. PROPELLANT ISOLATION VALVE (OXIDIZER)
9. PROPELLANT ISOLATION VALVE (FUEL)
10. TRANSDUCER HELIUM SPHERE TEMPERATURE
11. TRANSDUCER HELIUM SPHERE PRESSURE
12. TRANSDUCER FUEL TANK PRESSURE
13. TRANSDUCER OXIDIZER TANK PRESSURE
14. Δ P SWITCH FUEL/OXIDIZER
15. PRESSURE SWITCH OXIDIZER TANK
16. PRESSURE SWITCH FUEL TANK
17. 0.027 IN. DIAMETER ORIFICES

Fig. 2-37 Propellant Pressurization System

Propellant Vent Quick Disconnect – The airborne half of the propellant vent QD, when mated with the AGE half, permits access to the vehicle propellant tanks for pressurizing and venting operations without loss of propellant fumes or pressurization gas. After separation of the halves, the self-sealing airborne portion prevents the loss of pressurization gas and propellant fumes.

Construction – The propellant-vent quick disconnect (Fig. 2-38) and the high-pressure helium disconnect are similar in design. These units are of double-poppet design; one half is in the airborne portion and the other is the AGE portion of the assembly. Locking is accomplished by a ball, detent, and sleeve-type ball-retainer arrangement as described in Section 2.3.4. The QD is normally disconnected immediately at liftoff by a pneumatic actuator pulling the lanyard of the connector ground half. If the pneumatic actuator should fail, the lanyard itself, when brought into tension by vehicle liftoff, causes the ground half to disconnect. If both fail, a section of the airborne coupling shears (Fig. 2-38), which allows release without destroying the self-sealing capacity of the airborne coupling.

2.4.2 Helium Sphere

The helium sphere supplies high-pressure gas for pressurizing the propellant tanks during flight. Due to the extended orbital life requirements of the Gemini mission, a program peculiar 3653 $in.^3$ sphere is used instead of the standard 1612 $in.^3$ sphere, thereby adding an additional margin of safety and assuring the attainment of mission objectives. This sphere is mounted on a pedestal in the center of the forward section, as shown in Fig. 2-39. Specifications of the sphere are presented in Table 2-4.

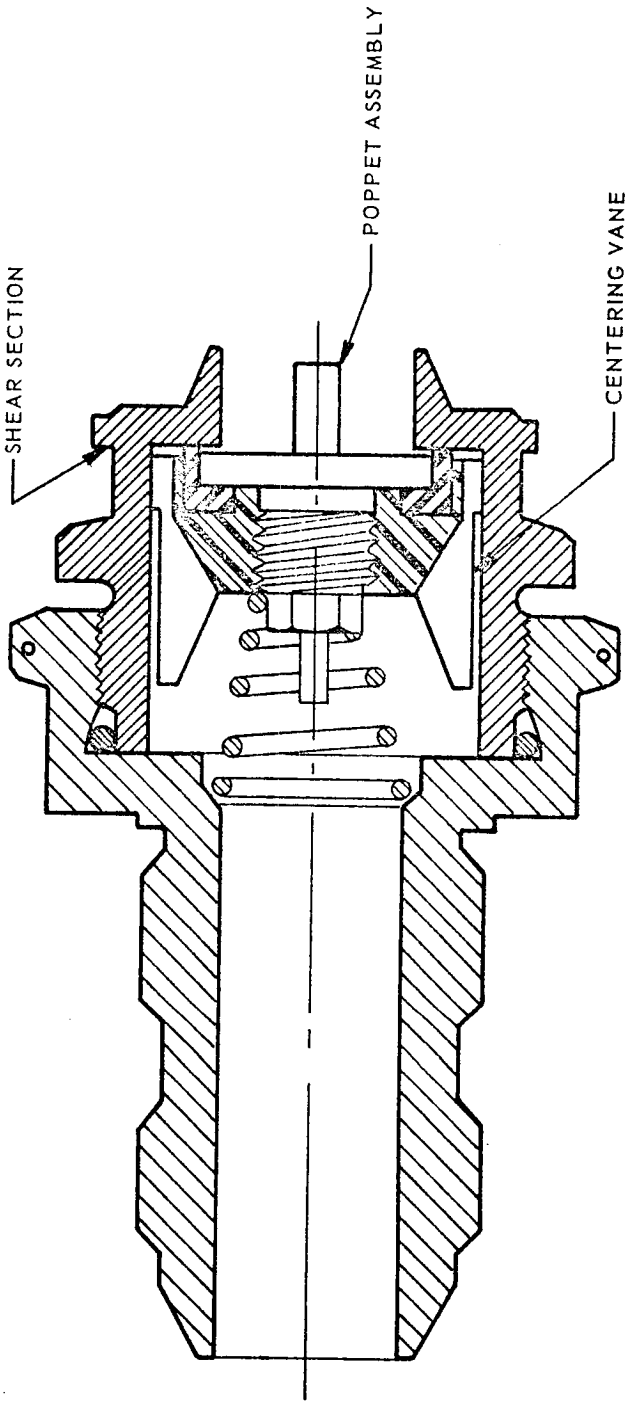


Fig. 2-38 Airborne Portion of Propellant Tank Vent Disconnect

Fig. 2-39 Helium Sphere Mounting Method

Table 2-4

HELIUM SPHERE SPECIFICATIONS

Volume	3653 In.^{3}	
Working Pressure	2500	psig
Proof Pressure	3000	psig
Burst Pressure	4000	psig
Mass of Helium (loaded at 120°F and 2500 psig)	3.144	lb

2.4.3 Pyrotechnically Operated Helium-Control Valve

Purpose – Propellant-tank pressure-time characteristics are controlled by the Pyrotechnically Operated Helium-Control Valve (POHCV). High-pressure gas for the initial pressurization of the propellant tanks is introduced through the vent couplings and passages of the POHCV prior to liftoff. During flight, actuation of the valve controls the introduction of pressurization gas from the helium sphere into the propellant tanks.

Figure 2-40 shows the normal or liftoff condition of the POHCV. In this mode, the helium sphere is filled through the helium-fill disconnect, and the propellant tanks are pressurized by means of the vent disconnects. At liftoff, the vent disconnects and the helium-fill disconnects self seal, and this mode is maintained until Agena first burn. Pressures in the helium sphere and propellant tanks are maintained entirely by the disconnect assemblies.

First Actuation – At 1.5 sec (nominal) after the engine-start signal, squibs 1 and 2 are fired (Fig. 2-41). High-pressure gas forces the piston to move to the left, thus isolating the helium-fill disconnect and allowing high-pressure helium to flow into the POHCV which actuates another piston (shown schematically as two). Actuation of the second piston isolates the vent coupling

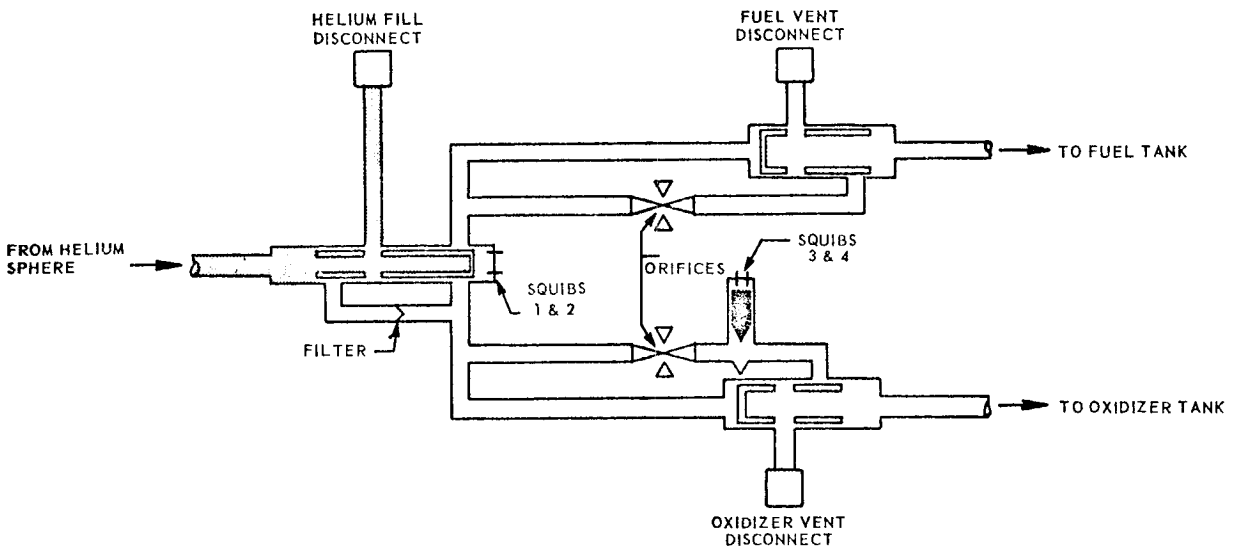


Fig. 2-40 Pyrotechnically Operated Helium-Control Valve – Normal Condition

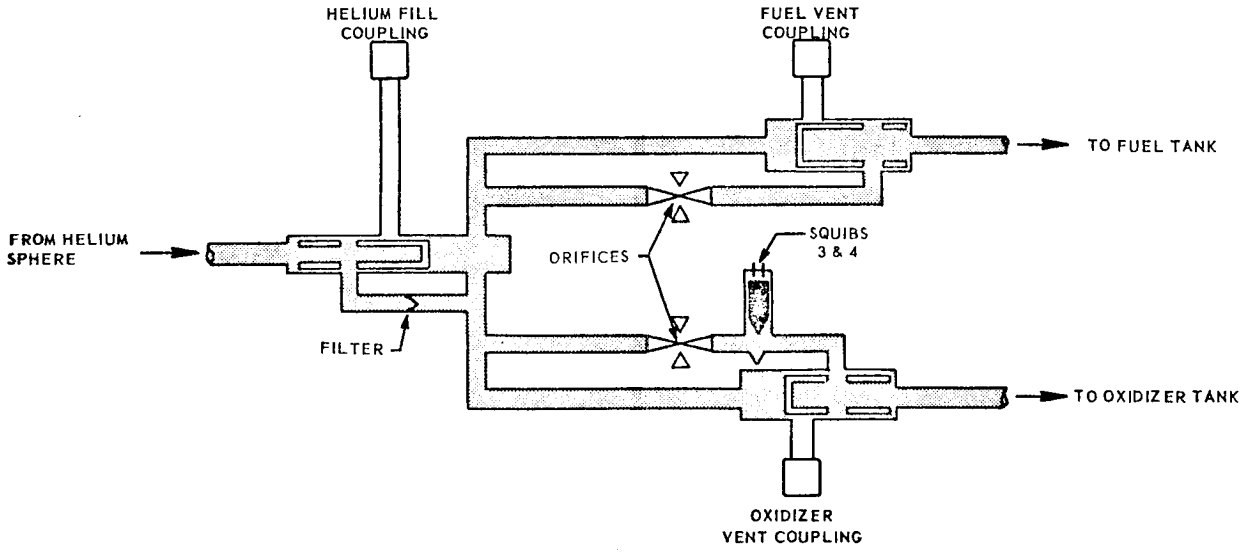


Fig. 2-41 Pyrotechnically Operated Helium-Control Valve – First Actuation

disconnects and opens up the gas passages from the sphere to the propellant tanks. At the same time, helium is applied across a pair of 0.027-in. diameter orifices which control the pressurization-gas flow to the propellant tanks. Figure 2-42 presents a typical pressure-time history for the sphere and propellant tanks.

Second Actuation – Approximately 318 sec after the engine-start signal, squibs 3 and 4 are fired, thus shutting off the pressurization-gas flow to the oxidizer tank (Fig. 2-42). The 318 sec is an optimum time which was established from extensive studies that considered the required pump-inlet pressures and the necessity of maintaining a 2.0 ±0.2-psid fuel-over-oxidizer tank pressure for the astronauts' safety.

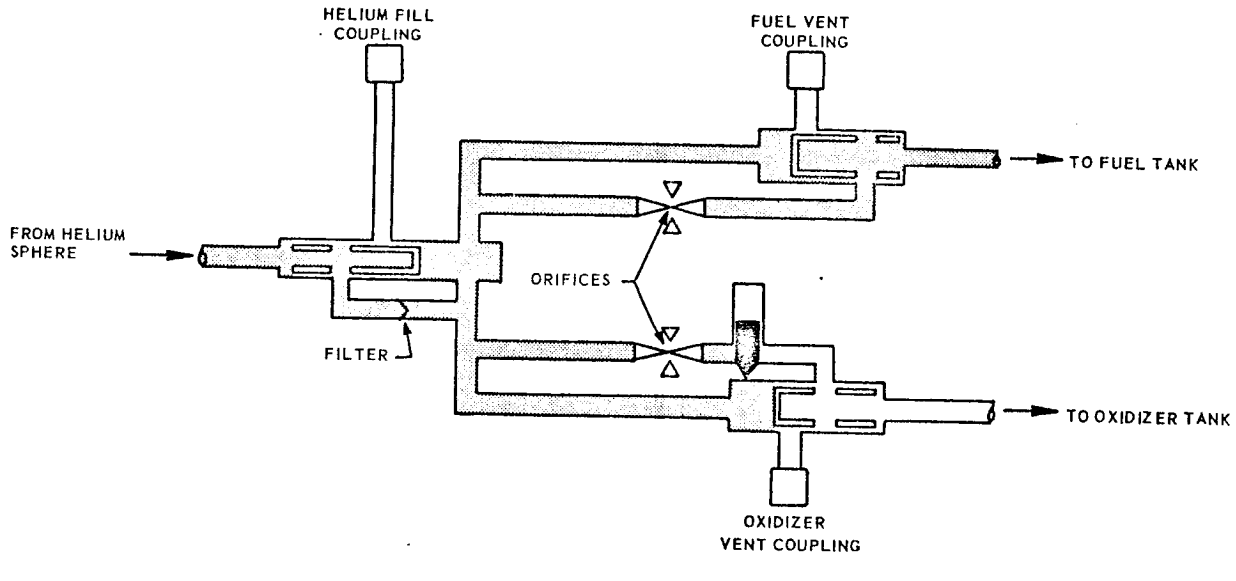


Fig. 2-42 Pyrotechnically Operated Helium-Control Valve – Second Actuation

Construction – Interport closure is accomplished with pressure (squib) actuated tapered pins which provide a leak-proof metal-to-metal seal. Ports are provided so that transducers may be mounted to monitor helium-sphere, fuel tank, and oxidizer tank pressures. A 10-micron nominal (25-micron absolute) filter is located between the pressurization (helium tank) port and orifices, downstream from the helium-fill coupling.

2.4.4 Tank-Mounted Pressure Switches

Fuel Tank Pressure Switch – The fuel tank pressure switch is mounted in the pressurization system and senses fuel tank-top pressure. The switch is set to actuate at 15 ±2 psia. This switch, although part of the pressurization system, does not in any way control the pressurization system and only serves to relate information for the ATV-mounted Status Display Panel. A full discussion of this panel is included in Section 5 of this report.

Oxidizer Tank Pressure Switch – The oxidizer tank pressure switch is mounted in the pressurization system and senses oxidizer tank-top pressure. The switch is set to actuate at 15 ±2 psia. This switch is similar to the fuel tank pressure switch and will be discussed in Section 5.

Propellant Tank ΔP Pressure Switch – The propellant tank ΔP switch is used to sense the relationship or difference in pressure between the fuel and oxidizer tanks. This switch is designed with separate dual-sensing chambers which are common only electrically. Like both of the previously mentioned pressure switches, this switch is used to supply information to the Status Display Panel and will be discussed in Section 5.

Section 3
PROPELLANTS AND PROPELLANT LOADINGS

3.1 ROCKET ENGINE PROPELLANTS

3.1.1 Fuel Propellant

Fuel utilized in the Gemini-Agena Target Vehicle (ATV) conforms to specification MIL-D-25604B-1 unsymmetrical dimethylhydrazine (UDMH). The fuel supplied to the fuel pump inlet may not contain particles greater than 0.030-in. in diameter; a filter located on the fuel pump prevents passage of particles greater than 0.025-in. in diameter to the gas generator plumbing.

Safety Considerations. Although UDMH vapor inhalation is the most dangerous hazard source, the fuel also can be absorbed into the body through contact with the skin and by ingestion. The effects of UDMH are severe irritation of mucous membranes of eyes, respiratory passages, and lungs; stimulation of the central nervous system resulting in convulsions; and irritation of the gastro-intestinal tract; nausea; and emesis. In sufficient quantities, the fuel can cause direct destruction of red blood cells.

Physical Properties. UDMH has a flash point of 34°F, is highly flammable, and is hypergolic in contact with IRFNA. Auto-ignition of the fuel occurs at approximately 482°F. UDMH liquid is subject to oxidation which causes the color to change from a clear to yellow. The fuel readily combines with moisture in the atmosphere which has a detrimental effect on its use as a rocket engine propellant. Spillage of the fuel may be neutralized with water and/or dilute sulfuric acid. Specific gravity of UDMH is shown in Fig. 3-1.

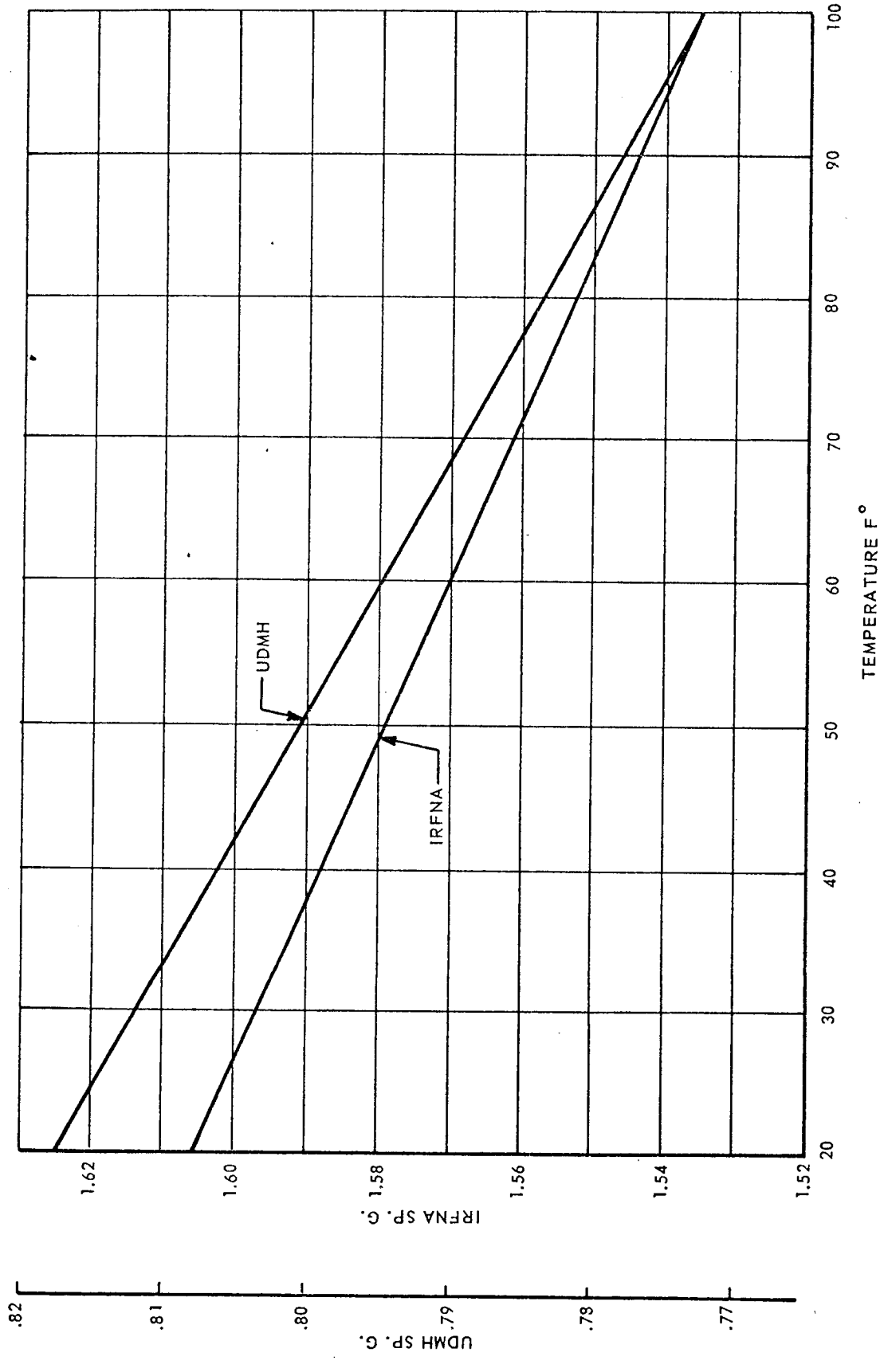


Fig. 3-1 Specific Gravity Versus Temperature (UDMH and IRFNA)

3.1.2 Oxidizer Propellant

Oxidizer used in the Gemini-ATV propulsion system conforms to Specification MIL-P-7254E-1, Type IIIB inhibited red fuming nitric acid (IRFNA). Total solids content of the oxidizer is less than 0.05 percent by weight. The IRFNA supplied to the oxidizer pump inlet may not contain particles greater than 0.40-in. in diameter; a filter located on the oxidizer pump prevents passage of particles greater than 0.010-in. in diameter to the gas generator plumbing.

Safety Considerations. IRFNA (both liquid and vapor) is extremely toxic in high concentrations. Breathing the vapor may cause severe damage to the nose, throat, respiratory passages, and lungs. In contact with the skin or eyes, nitric acid causes severe burning; in the case of the eyes, possible destruction of the eye tissues can occur with subsequent blindness. The oxidizer also vigorously attacks most metal and organic materials and in concentrated form may cause spontaneous ignition and vigorous burning when in contact with organic materials.

Physical Properties. IRFNA has a boiling point of 142°F and a freezing point of -65°F and is extremely corrosive. The oxidizer is hypergolic when combined with UDMH. Specific gravity of this acid is shown in Fig. 3-1.

3.2 OTHER MEDIUMS

In addition to propellants, other mediums used by the Primary Propulsion System (PPS) are helium for propellant tank pressurization, nitrogen for start-tank pressurization, and a nitrogen-freon mixture for oxidizer-pump lip-seal pressurization.

3.2.1 Helium

The helium used for propellant tank pressurization conforms to U.S. Bureau of Mines Grade A specification. Moisture content of the gas introduced into the propulsion system is less than 26.3 parts per million (ppm) by volume at standard conditions. Cleanliness requirements are shown in Table 3-1.

3.2.2 Nitrogen

Nitrogen used for pressurizing the start tank bellows conforms to Federal Standard BB-N-411, Type I, Class I, Grade B. The moisture content is less than 26.3 ppm by volume under standard conditions. The cleanliness requirements are presented in Table 3-1.

Table 3-1

CLEANLINESS REQUIREMENTS OF PNEUMATIC FLUID MEDIA

Particle Size (microns)	Number Permitted	
	Metallic	Non-Metallic
26 to 50	100	1000
51 to 100	30	300
101 to 150	0	50
Over 150	0	0

3.2.3 Nitrogen Freon-14 Mixture

The nitrogen freon-14 mixture used for pressurizing the lip seal is supplied from a small regulator through a normally closed-solenoid valve in the vehicle attitude-control system. This solenoid valve is energized to open any time that liquid is present in the Model 8247 engine system.

At AMR, the gas is purchased as a mixture with the two gases meeting the following specifications: nitrogen conforming to Federal Standard BB-N-411 Type I, Class I, Grade B; freon conforming to MS-31-4000, tetrafluoromethane (freon-14). The mixture has a moisture content less than 20 ppm by volume under standard atmospheric conditions. Cleanliness requirements of the mixture are shown in Table 3-1.

3.3 PROPELLANT LOADING

3.3.1 Method of Propellant Loading

Propellant tanks are loaded with fuel and oxidizer during the final hours before vehicle launching. The propellants are pumped through the launch pad propellant-transfer equipment and fill lines (in the umbilical mast) to the ground quick-disconnect couplings. These ground couplings attach to the airborne halves of the vehicle fuel and oxidizer fill couplings. Vehicle propellant tanks are also equipped with vent lines that terminate at airborne quick-disconnect couplings mounted near the forward end of the propellant tank assembly. The vehicle vents are coupled to return lines so that a closed-loop circuit is formed by the propellant-transfer equipment, load lines, propellant tanks, and vent lines. By means of this closed-loop arrangement, the normal process of venting the tanks during propellant loading can be reversed and, by means of pressurization, drain the tanks of propellants, if necessary.

3.3.2 Propellant Loading Calculations (Main Tanks)

The loading of the main propellant tanks for flight is computed as outlined in this section.

Tables 3-2 and 3-3 present the breakdown of non-impulse and transient-impulse propellants for the burns as indicated. Residual, or trapped non-impulse propellants, are shown in Table 3-4. The nominal propellant tank volume available for propellant loading is shown in Table 3-5.

Loading calculations are based on actual calibrated tank volumes for each vehicle considered; the vehicle log book contains the calibrated volumes for the particular vehicle. Nominal tank volumes (including sumps) are 77.45 ft^3 for fuel and 100.50 ft^3 for oxidizer.

Table 3-2

TRANSIENT- AND NON-IMPULSE PROPELLANTS INITIAL BURNS

Non-Impulse Propellants	Oxidizer (lb)	Fuel (lb)
Pre-flow		
Propellant flow past injector prior to thrust-chamber ignition	0.00	2.00
Cooling Passages		
Propellant expelled after each shutdown	11.50	0.00
Propellant Isolation Valves		
Propellant expelled after each shutdown	8.69	3.93
Propellant flow from fuel valve closure to oxidizer valve closure	19.80	0.00
Total	39.99	5.93
Transient-Impulse Propellants		
Propellant flow from 0 to 75% P_c	1.58	0.56
Residual propellant in start tanks	0.17	0.86
Propellant flow from shutdown signal to fuel-valve closure	1.15	0.45
Total	2.90	1.87
Total transient- and non-impulse propellants for each initial burn	42.89 = A	7.80 = B

Table 3-3

TRANSIENT- AND NON-IMPULSE PROPELLANTS FINAL BURN
(Assuming Propellant Depletion)

Non-Impulse Propellants	Oxidizer (lb)	Fuel (lb)
Pre-flow		
Propellant flow past injector prior to thrust chamber ignition	0.00	2.00
Cooling Passages		
Propellant expelled after final shutdown	10.50	0.00
Total	10.50	2.00
Transient-Impulse Propellants		
Residual propellant in start tanks	0.17	0.86
Propellant flow from 0 to 75% P_c	1.58	0.56
Propellant flow from 70 to 0% P_c	8.20	0.0
Total	9.95	1.42
Total transient- and non-impulse propellants for final burn	20.45 = C	3.42 = D

Table 3-4

RESIDUALS (TRAPPED NON-IMPULSE PROPELLANTS)

	Oxidizer (lb)	Fuel (lb)
Feed lines (including PIV's)	2.3	1.6
Propellant vapor (tanks at 60°F and 10 to 12 psia)	1.8	1.2
Fill lines	1.5	0.8
Fuel stand pipe	0.0	0.9
Pump to engine propellant valves	5.2	1.7
Fuel bias	0.0	12.0
	10.8 = E	18.2 = F

Table 3-5

NOMINAL TANK VOLUMES

	Oxidizer (ft^3)	Fuel (ft^3)
Nominal Tank Volumes*	100.50	77.45
Feed, fill, and engine	0.09	0.08
Tank stretch	0.05	0.08
Total Volume	100.64	77.61
Minimum ullage	-0.90	-0.75
Nominal volume available for propellant loading	99.74	76.86

*These are nominal values. Actual tank volumes on individual vehicles will be accurately calibrated.

The fuel tank is loaded first to the minimum ullage value and then, using the altitude-corrected engine-system-mixture ratio, the oxidizer tank load is calculated. The oxidizer tank is checked to verify that loading does not exceed the minimum ullage value for the tank. If the loading into the oxidizer tank exceeds this minimum ullage value, the process is reversed; the oxidizer tank is loaded first, and then, using the corrected mixture ratio, the fuel tank load is calculated.

Loading of either tank must be predicated upon the mission profile which indicates the number of PPS burns to be attempted. After each shutdown of the Model 8247 engine, varying quantities of oxidizer are vented, and due to the dissimilarity of these amounts, the loading must necessarily vary with the number of burns to be attempted. In the following example arbitrarily chosen, propellant tank loading equations and typical calculations for a mission profile of 5 PPS burns are presented.

Fuel Tank Loading Equations

a. Equation 1

$$V_{fl} = V_f - V_{uf}$$

where:

V_{fl} = volume of fuel to be loaded (ft^3)

V_f = calibrated volume of fuel tank (ft^3 – obtained from vehicle log book and Table 3-5)

V_{uf} = minimum fuel tank ullage. Constant 0.75 ft^3

Note: This equation produces the same values as Table 3-5

b. Equation 2

$$W_{fl} = (V_{fl})(\rho_{fn})$$

where:

W_{fl} = weight of fuel to be loaded
ρ_{fn} = density of fuel at $60^{\circ}F = 49.582$ lb/ft^3

Oxidizer Tank Loading Equations

a. Equation 3

$$W_{fi} = W_{fl} - C_1$$

where:

W_{fi} = weight of impulse fuel (lb)
C_1 = constant total weight of non-impulse fuel, defined as follows:

$$C_1 = (M-1)(B) + D + F$$

where:

M = number of engine burns in mission
B = transient- and non-impulse propellants for initial burns (Table 3-2)
D = transient- and non-impulse propellants for final burn (Table 3-3)
F = residual propellants (Table 3-4)

b. Equation 4

$$W_{oi} = (W_{fi})(R_{\infty})$$

where:

W_{oi} = weight of impulse oxidizer (lb)

R_{∞} = total engine system mixture ratio at altitude. Sample calculation shown in Paragraph 6.8.

c. Equation 5

$$W_{ol} = W_{oi} + C_2$$

where:

W_{ol} = weight of oxidizer to be loaded (lb)

C_2 = constant, total weight of non-impulse oxidizer defined as follows:

$$C_2 = (M-1)(A) + C + E$$

where:

A = transient- and non-impulse propellants for initial burns (Table 3-2)

C = transient- and non-impulse propellants for final burn (Table 3-3)

E = residual propellants (Table 3-4)

Check Oxidizer Tank Ullage

a. Equation 6

$$V_{ol} = \frac{(W_{ol})}{\rho_{on}}$$

where:

ρ_{on} = density of oxidizer at $60^{\circ}F$ = 97.916 lb/ft^3

V_{ol} = volume of oxidizer loaded

b. Equation 7

$$V_{uo} = V_o - V_{ol}$$

V_o = calibrated volume of oxidizer tank (ft^3 — obtained from vehicle log book and Table 3-5)

V_{uo} = minimum oxidizer tank ullage. Constant - 0.90 ft^3

$V_{uo} \geqq 0.90\ ft^3$ for correct loading

Example Calculation

V_f = 77.21 (vehicle log book and Table 3-5)

V_o = 100.64 (vehicle log book and Table 3-5)

R = 2.551

M = 5

Fuel

$$V_{fl} = V_f - V_{uf} \qquad \text{(Eq. 1)}$$
$$V_{fl} = 77.21 - 0.75$$
$$V_{fl} = 76.46\ ft^3$$

$$W_{fl} = V_{fl}(\rho_{fn}) \qquad \text{(Eq. 2)}$$
$$W_{fl} = (76.46)(49.582)$$
$$W_{fl} = 3791.04\ lb$$

Oxidizer

$$W_{fi} = W_{fl} - C_1 \qquad \text{(Eq. 3)}$$

$$W_{fi} = 3791.04 - \left[(4)(7.80) + 3.42 + 18.20\right]$$

$$W_{fi} = 3738.22 \text{ lb}$$

$$W_{oi} = (W_{fi})(R_\infty) \qquad \text{(Eq. 4)}$$

$$W_{oi} = (3738.22)(2.551)$$

$$W_{oi} = 9536.20 \text{ lb}$$

$$W_{ol} = W_{oi} + C_2 \qquad \text{(Eq. 5)}$$

$$W_{ol} = 9536.20 + \left[(4)(42.89) + 20.45 + 10.80\right]$$

$$W_{ol} = 9739.01 \text{ lb}$$

Ullage Check – Oxidizer

$$V_{ol} = \frac{W_{ol}}{\rho_{on}} \qquad \text{(Eq. 6)}$$

$$V_{ol} = \frac{9739.01}{97.916}$$

$$V_{ol} = 99.46 \text{ ft}^3$$

$$V_{uo} = V_o - V_{ol} \qquad \text{(Eq. 7)}$$

$$V_{uo} = 100.64 - 99.46$$

$$V_{uo} = 1.18 \text{ ft}^3$$

The actual ullage volume is greater than 0.90 ft^3; therefore, the loading is acceptable.

In the aforementioned example, in which fuel was loaded first, V_{uo} was acceptable. Frequently, due to variations in tank volumes and mixture ratio, V_{uo}, will not be sufficient when calculated by this method. In such cases, it is necessary to load the oxidizer first, substituting oxidizer parameters for fuel parameters in the existing equations. The following is an example of such a condition:

Example Calculation:

$$\left.\begin{aligned} V_f &= 77.61 \\ V_o &= 100.64 \end{aligned}\right\} \text{(vehicle log book and Table 3-5)}$$

$$R = 2.551$$

$$M = 5$$

Fuel:

$$V_{fl} = V_f - V_{uf} \qquad \text{(Eq. 1)}$$

$$V_{fl} = 77.61 - 0.75$$

$$V_{fl} = 76.86 \text{ ft}^3$$

$$W_{fl} = (V_{fl})(\rho_{fn}) \qquad \text{(Eq. 2)}$$

$$W_{fl} = (76.86)(49.582)$$

$$W_{fl} = 3810.87 \text{ lb}$$

Oxidizer:

$$W_{fl} = W_{fl} - C_1 \qquad \text{(Eq. 3)}$$

$$W_{fi} = 3810.87 - \left[(4)(7.80) + 3.42 + 18.20\right]$$

$$W_{fi} = 3758.05 \text{ lb}$$

$$W_{oi} = W_{fi}(R_\infty) \qquad \text{(Eq. 4)}$$

$$W_{oi} = (3758.05)(2.551)$$

$$W_{oi} = 9586.78 \text{ lb}$$

$$W_{ol} = W_{oi} + C_2 \qquad \text{(Eq. 5)}$$

$$W_{ol} = 9586.78 + \left[(4)(42.89) + 20.45 + 10.80\right]$$

$$W_{ol} = 9789.59 \text{ lb}$$

Ullage Check – Oxidizer:

$$V_{ol} = \frac{W_{ol}}{\rho_{on}} \qquad \text{(Eq. 6)}$$

$$V_{ol} = \frac{9789.59}{97.916}$$

$$V_{ol} = 99.98 \text{ ft}^3$$

$$V_{uo} = V_o - V_{ol} \qquad \text{(Eq. 7)}$$

$$V_{uo} = 100.64 - 99.98$$

$$V_{uo} = 0.66 \text{ ft}^3$$

The actual ullage volume is less than 0.90 ft^3; therefore, the loading is unacceptable.

It is necessary to recalculate the loading, starting with the oxidizer. The procedure is as follows:

Oxidizer:

$$V_{ol} = V_o - V_{uo} \qquad \text{(Eq. 1)}$$

$$V_{ol} = 100.64 - 0.90$$

$$V_{ol} = 99.74 \text{ ft}^3$$

$$W_{ol} = (V_{ol})(\rho_{on}) \qquad \text{(Eq. 2)}$$

$$W_{ol} = (99.74)(97.916)$$

$$W_{ol} = 9766.14 \text{ lb}$$

Fuel:

$$W_{oi} = W_{ol} - C_2 \qquad \text{(Eq. 3)}$$

$$W_{oi} = 9766.14 - \left[(4)(42.89) + 20.45 + 10.80\right]$$

$$W_{oi} = 9563.33 \text{ lb}$$

$$W_{if} = \frac{W_{oi}}{R_\infty} \qquad \text{(Eq. 4)}$$

$$W_{if} = \frac{9563.33}{2.551}$$

$$W_{if} = 3748.86 \text{ lb}$$

$$W_{fl} = W_{if} + C_1 \qquad \text{(Eq. 5)}$$

$$W_{fl} = 3748.86 + \left[(4)(7.80) + 3.42 + 18.20\right]$$

$$W_{fl} = 3801.68 \text{ lb}$$

Ullage Check – Fuel:

$$V_{fl} = \frac{W_{fl}}{\rho_{fn}} \qquad \text{(Eq. 6)}$$

$$V_{fl} = \frac{3801.68}{49.582}$$

$$V_{fl} = 76.67 \text{ ft}^3$$

$$V_{uf} = V_f - V_{fl} \qquad \text{(Eq. 7)}$$

$$V_{uf} = 77.61 - 76.67$$

$$V_{uf} = 0.94 \text{ ft}^3$$

The actual fuel ullage volume is greater than 0.75 ft^3; therefore, the loading is acceptable.

3.3.3 Start Tank Charging

Servicing of the Model 8247 engine start system is accomplished with two AGE servicing carts. These carts supply propellants filtered to 25 microns absolute and start tank charging N_2-gas filtered to 10 microns absolute.

Due to the critical nature of start tank loading, the method of loading is presented below:

3.3.3.1 Fuel Start Tank Loading

Servicing the Fuel Loading Cart. The fuel start tank loading cart is filled from the launch pad fuel bulk stores. A minimum of 130 $in.^3$ of UDMH is loaded into the cart. A sample of the fuel is then taken from the system and analyzed for conformity to MIL-D-25604B-1. The oxidizer and fuel nitrogen-charging bottles located in the fuel loading cart are pressurized to approximately 1500 psig.

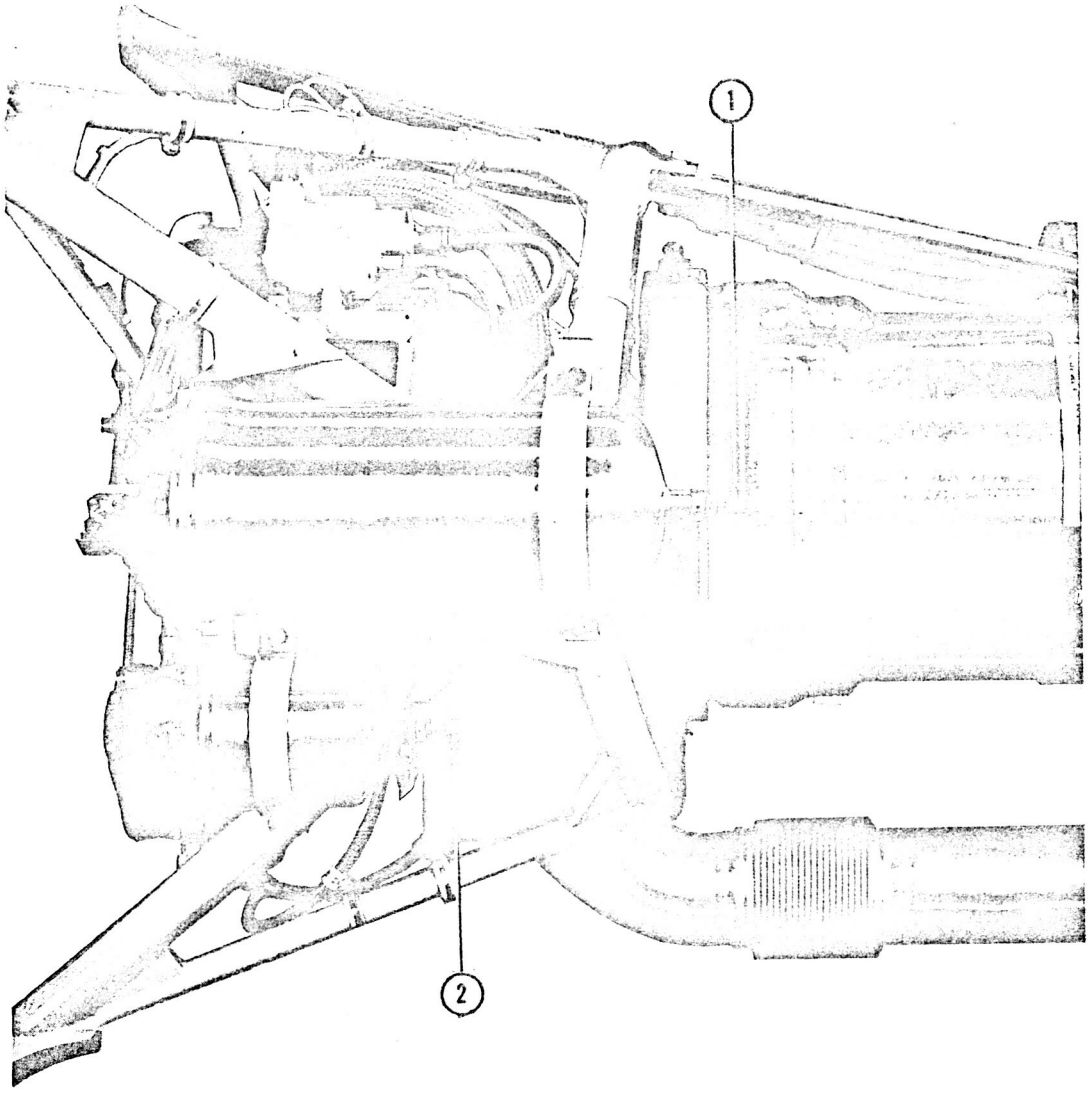
Fuel Start Tank Loading. The gas side of the start tank is pressurized (Item 1, Fig. 3-2) to stack the bellows, and the start tank-mounted liquid bleed valve (Item 1, Fig. 3-3) is opened – thereby forcing out trapped gas or liquid from the liquid side of the tank. The bleed valve is closed and the AGE liquid-transfer line is connected to the start tank fill valve (Item 2, Fig. 3-3). The volume to be transferred to the start tank is variable and must be determined from Table 3-6.

Table 3-6

QUANTITY OF UDMH TO BE LOADED INTO FUEL START TANK

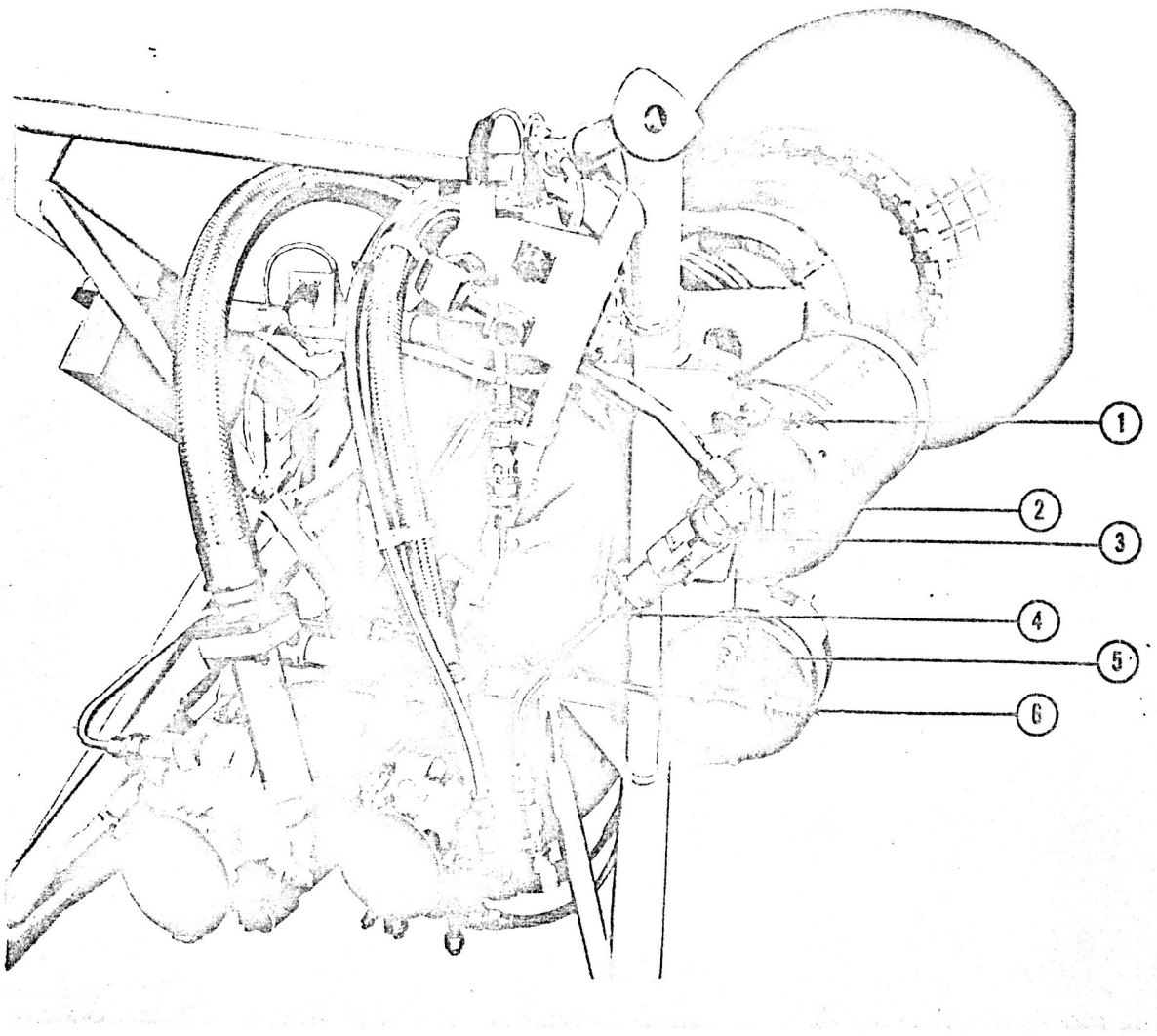
Fuel Temperature (°F)	Dry Load* Volume (in.^{3})	Wet Load** Volume (in.^{3})
40 ± 5	106.1 ± 0.9	$93.5^{+2.0}_{-0}$
50 ± 5	106.9 ± 0.9	$94.2^{+2.0}_{-0}$
60 ± 5	107.7 ± 0.9	$95.0^{+2.0}_{-0}$
70 ± 5	108.5 ± 0.9	$95.8^{+2.0}_{-0}$
80 ± 5	109.3 ± 0.9	$96.6^{+2.0}_{-0}$
90 ± 5	110.0 ± 0.9	$97.4^{+2.0}_{-0}$
100 ± 5	111.0 ± 0.9	$98.2^{+2.0}_{-0}$
110 ± 5	111.8 ± 0.9	$99.1^{+2.0}_{-0}$

*A fuel loading for a dry system includes the fuel necessary to completely fill an empty start tank bellows, the fuel system lines (to the gas generator solenoid valve and dual check valve), and instrumentation.
**A fuel loading for a wet system does not include the start tank residual, fuel system lines, and instrumentation volumes.



1 Fuel Start Tank Nitrogen Fill Port FP-11
2 Oxidizer Start Tank Nitrogen Fill Port FP-10

Fig. 3-2 Fuel and Oxidizer Start Tank Nitrogen Fill Ports



1 Fuel Start Tank Valve Bleed Point BP-4
2 Fuel Start Tank Fill Port FP-6
3 Fuel Dual Check Valve Outlet TP-14
4 Fuel Dual Check Valve Inlet TP-16
5 Oxidizer Start Tank Valve Bleed Point BP-5
6 Oxidizer Start Tank Fill Port FP-9

Fig. 3-3 Engine Start Tank System

3.3.3.2 Oxidizer Start Tank Loading

Servicing the Oxidizer Loading Cart. The oxidizer start tank loading cart is filled from the launch pad oxidizer bulk stores. A minimum of 35 $in.^3$ of oxidizer is drawn into the sight glass, and all AGE plumbing is bled in order to eliminate the possibility of loading trapped voids into the engine start tank. A sample of the oxidizer is then taken from the system and analyzed for conformity to MIL-P-7254-1 Type IIIB.

Oxidizer Start Tank Loading. The gas side of the start tank is pressurized (Item 2, Fig. 3-2) to stack the bellows, and the start tank-mounted liquid-bleed valve (Item 5, Fig. 3-3) is opened — thereby forcing out trapped gas or liquid from the liquid side of the tank. The bleed valve is closed and the AGE liquid-transfer line is connected to the start tank fill valve (Item 6, Fig. 3-3). The volume to be transferred to the start tank is variable and must be determined from Table 3-7.

3.3.3.3 Nitrogen Precharging of Start Tanks

Fuel Tank. The fuel nitrogen-charge tank (located on the fuel start tank service cart) is charged with gas at 1480 to 1500 psig. This charging tank is connected to the start tank at FP-11 (Item 1, Fig. 3-2). Using the UDMH temperature in the start tank and the gas temperature in the nitrogen charge tank, the proper charging pressure is determined from the appropriate curve (Figs. 3-4a through 3-4h). The charging tank is bled slowly (so as not to reduce gas temperature and change charging requirements) to pressure "A", pressurizing the start tank bellows to the limits of pressure "B". The gas charging line is now removed, and all ports are pressure capped.

Oxidizer Tank. The oxidizer nitrogen-charge tank (located on the fuel start tank service cart) is charged with gas at 1325 to 1350 psig. This charging

tank is connected to the start tank at FP-10 (Item 2, Fig. 3-2). Using the liquid temperature in the start tank and the gas temperature in the nitrogen charge tank, the proper charging pressure is determined from the appropriate curve (Figs. 3-5a through 3-5h). The charging tank is bled slowly (so as not to reduce gas temperature and change charging requirements) to pressure "A", charging the start tanks to the limits of pressure "B". The gas charging line is now removed, and all ports are pressure capped.

Table 3-7

QUANTITY OF IRFNA TO BE LOADED INTO OXIDIZER-START TANK

Oxidizer Temperature (^{o}F)	Dry Load* Volume (in.^{3})	Wet Load** Volume (in.^{3})
40 ± 5	17.5 ± 0.16	$13.6^{+0.5}_{-0}$
50 ± 5	17.6 ± 0.16	$13.7^{+0.5}_{-0}$
60 ± 5	17.7 ± 0.16	$13.8^{+0.5}_{-0}$
70 ± 5	17.8 ± 0.16	$13.9^{+0.5}_{-0}$
80 ± 5	17.9 ± 0.16	$14.0^{+0.5}_{-0}$
90 ± 5	18.0 ± 0.16	$14.1^{+0.5}_{-0}$
100 ± 5	18.1 ± 0.16	$14.2^{+0.5}_{-0}$
110 ± 5	18.2 ± 0.16	$14.3^{+0.5}_{-0}$

*An oxidizer loading for a dry system includes the oxidizer necessary to completely fill an empty start tank bellows, the oxidizer system lines (to the gas generator solenoid valve and dual check valve), and instrumentation.
**An oxidizer loading for a wet system does not include the start tank residual, oxidizer system lines, and instrumentation volumes.

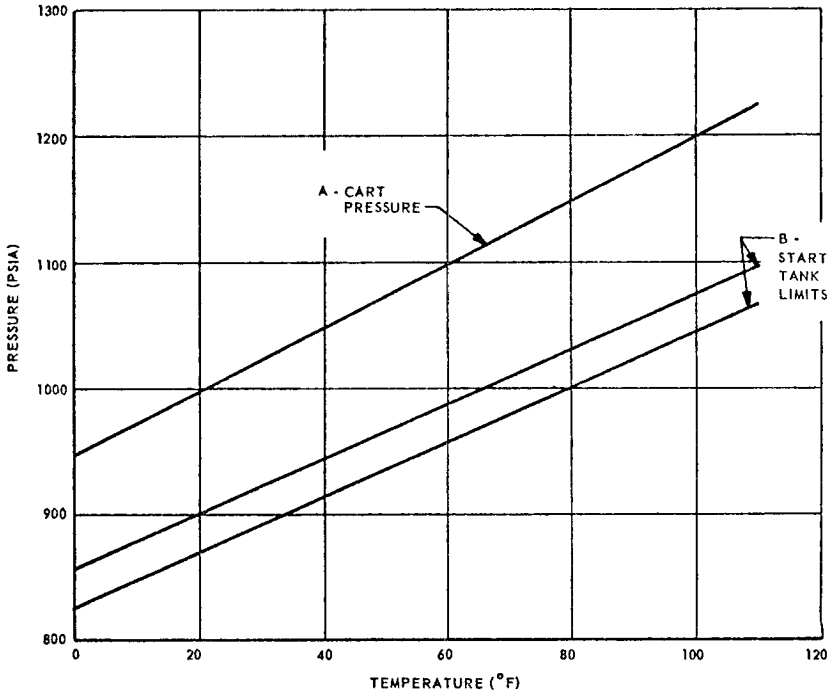


Fig. 3-4a Fuel To Be Loaded at 40° ±5°F

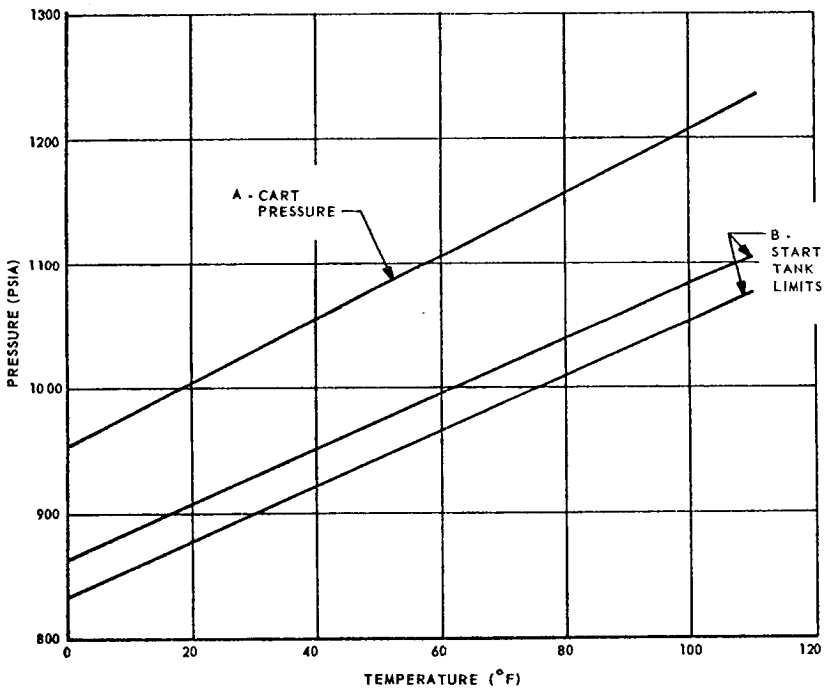


Fig. 3-4b Fuel To Be Loaded at 50° ±5°F

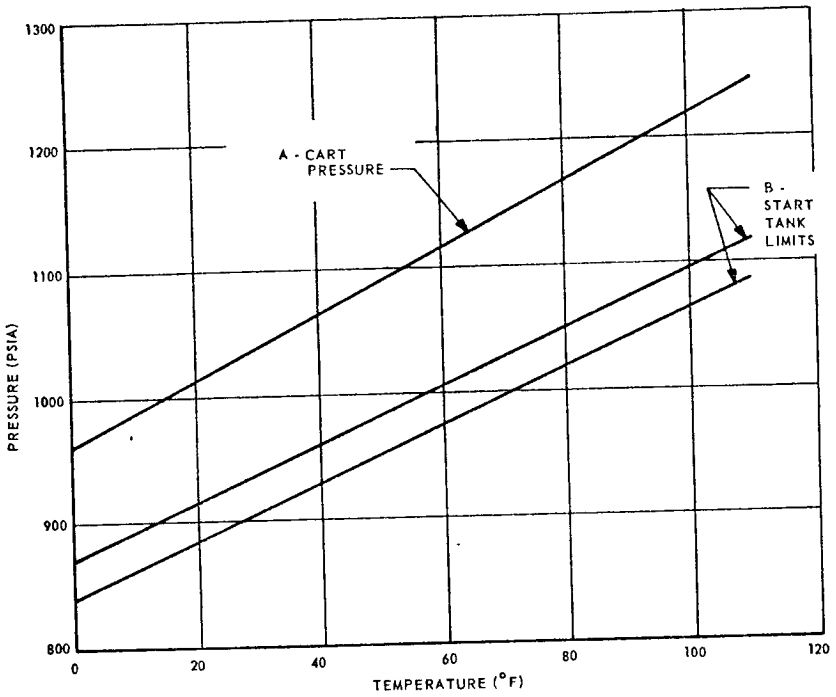


Fig. 3-4c Fuel To Be Loaded at 60° ±5°F

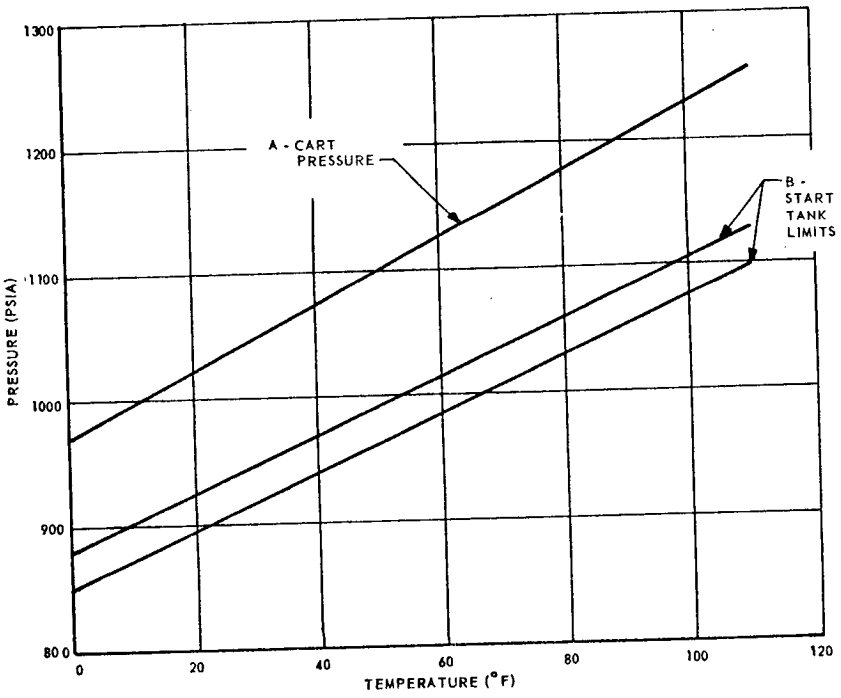


Fig. 3-4d Fuel To Be Loaded at 70° ±5°F

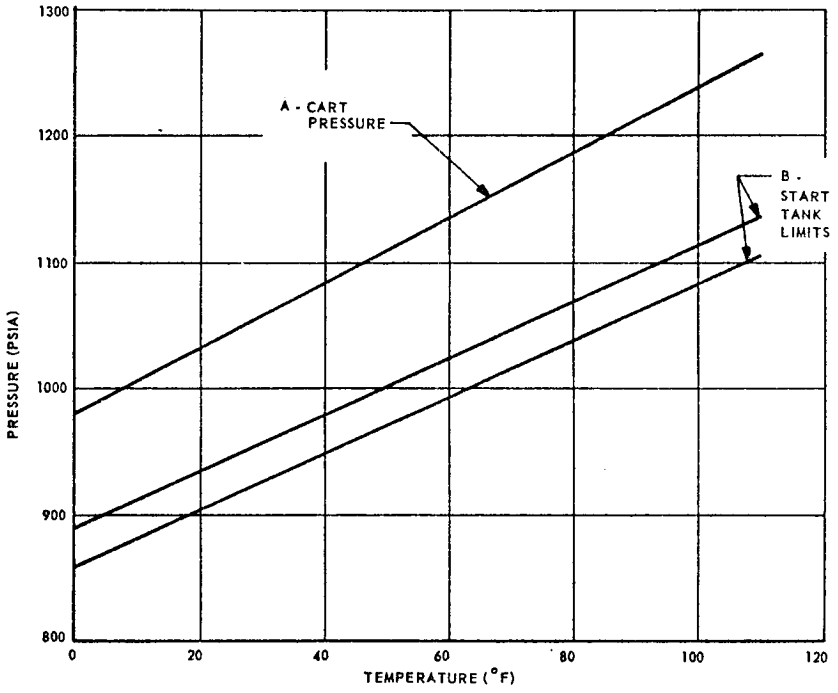


Fig. 3-4e Fuel To Be Loaded at 80° ±5°F

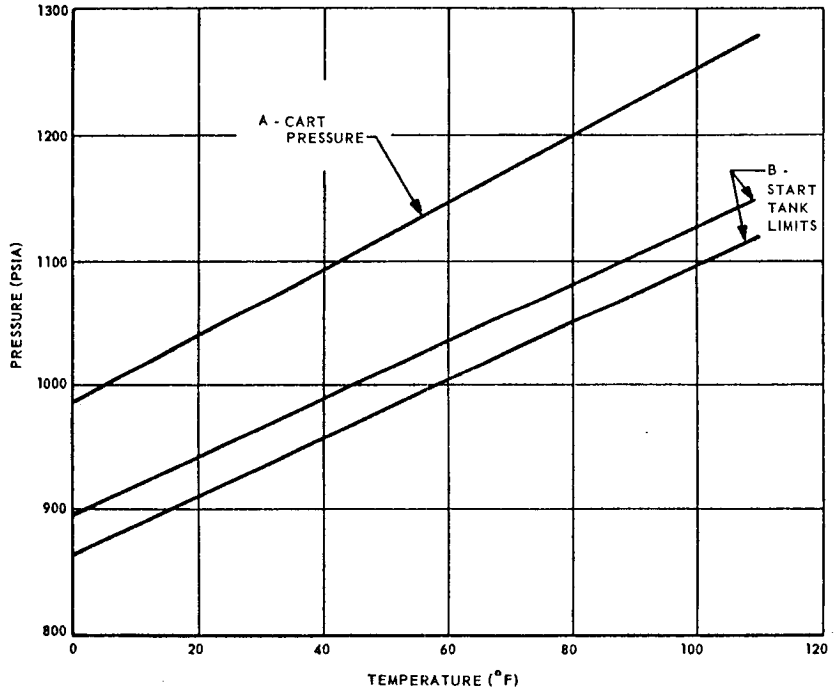


Fig. 3-4f Fuel To Be Loaded at 90° ±5°F

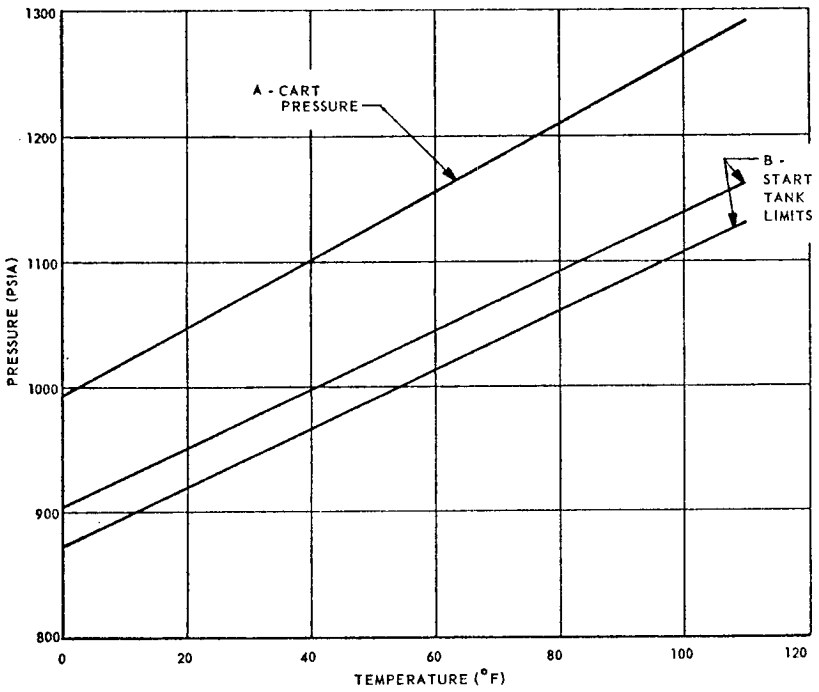


Fig. 3-4g Fuel To Be Loaded at 100° ±5°F

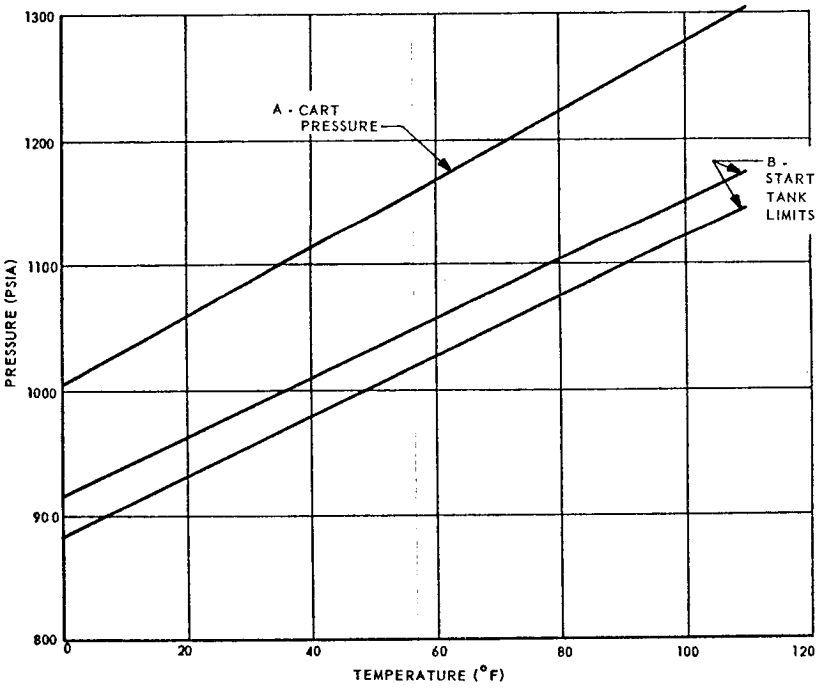


Fig. 3-4h Fuel To Be Loaded at 110° ±5°F

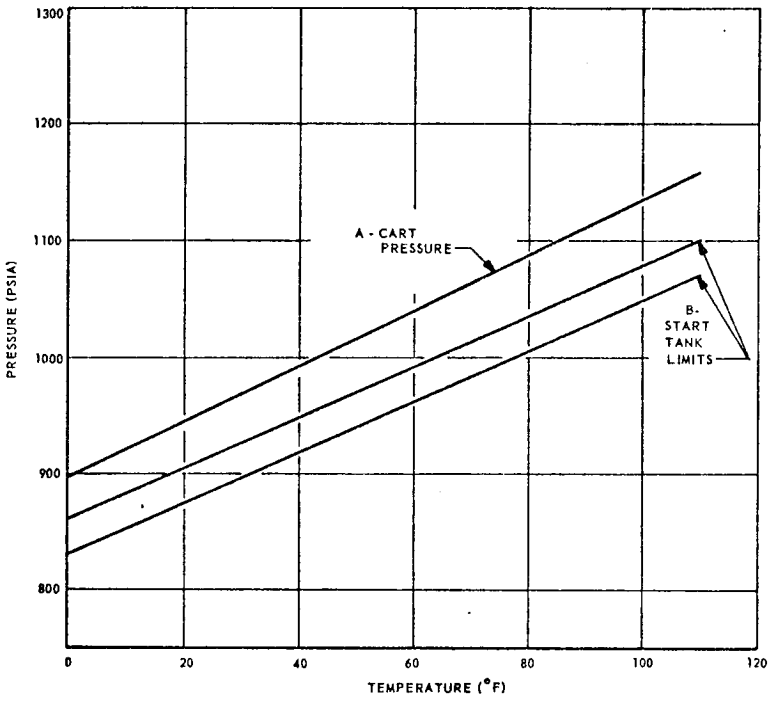


Fig. 3-5a Oxidizer To Be Loaded at 40° ±5°F

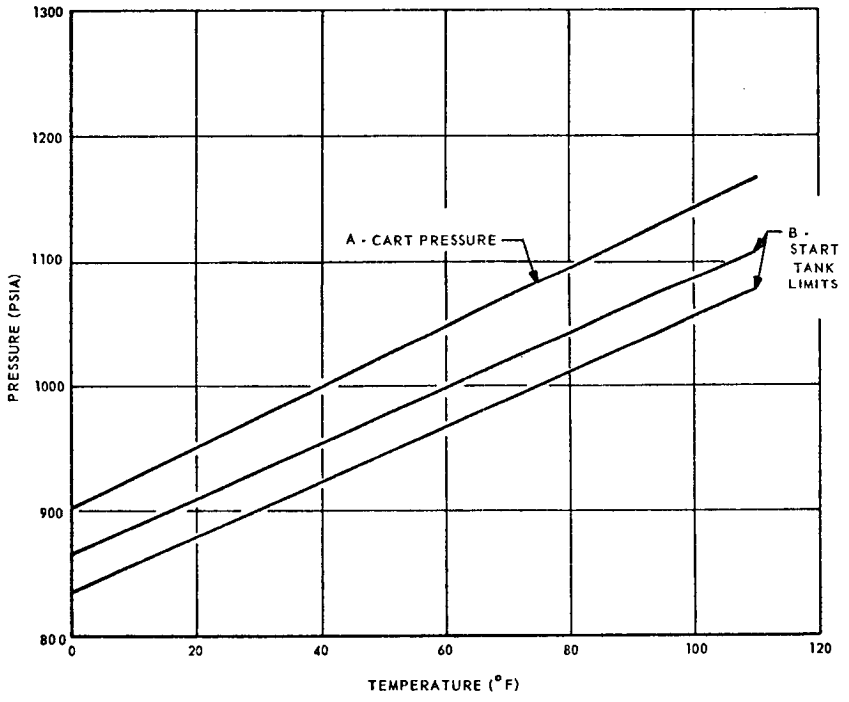


Fig. 3-5b Oxidizer To Be Loaded at 50° ±5°F

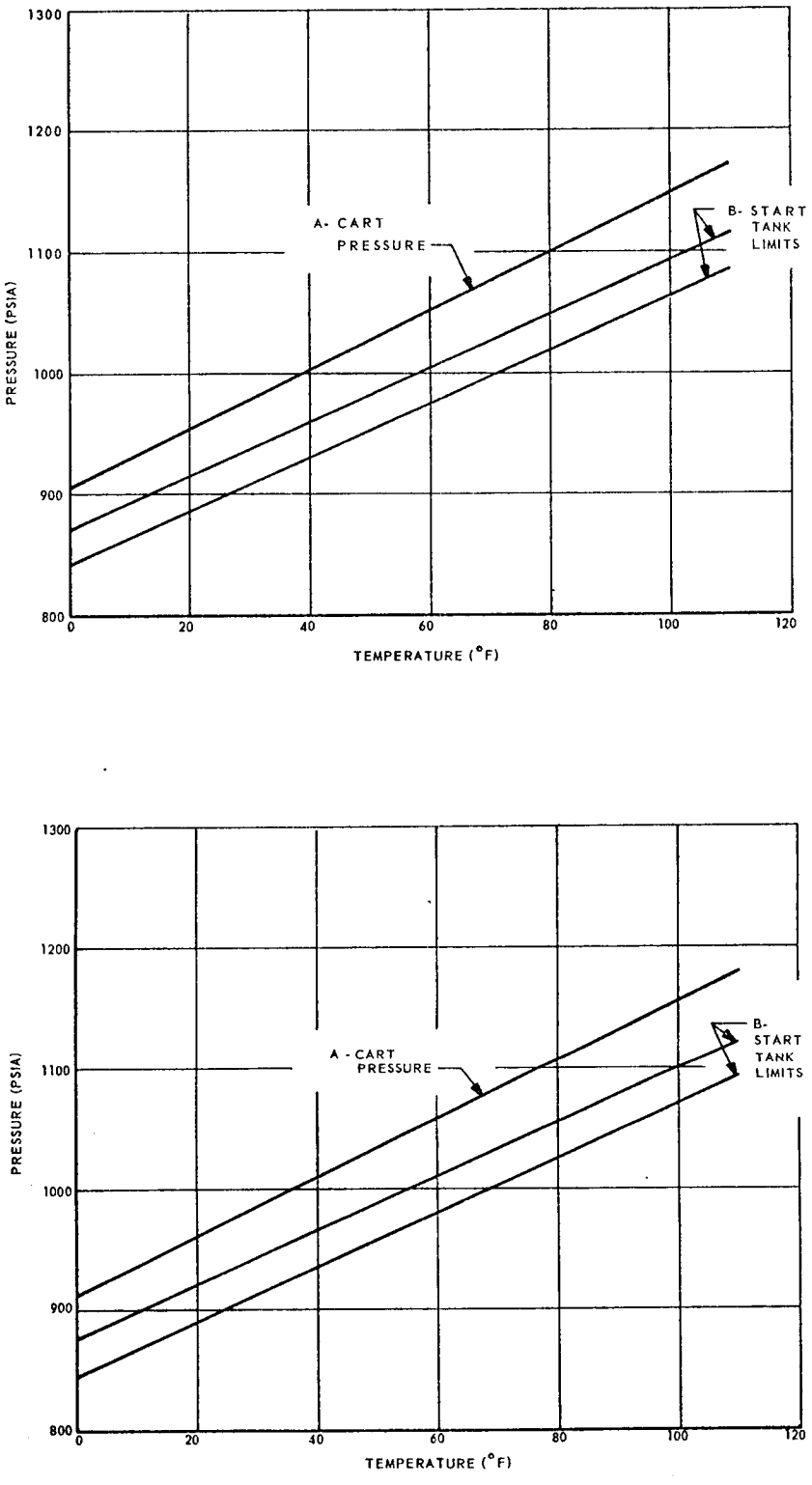


Fig. 3-5d Oxidizer To Be Loaded at 70° ±5°F

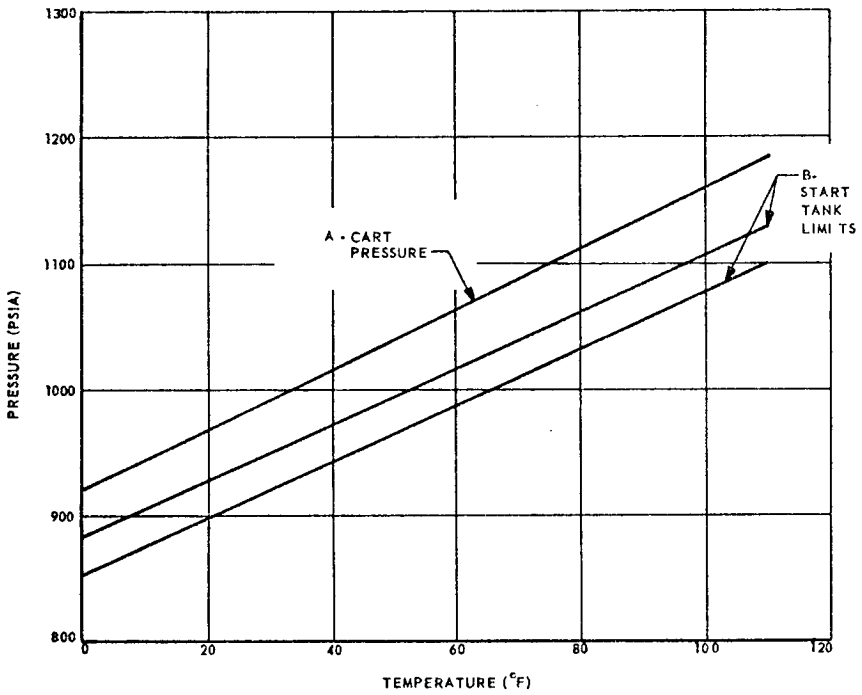


Fig. 3-5e Oxidizer To Be Loaded at 80° ±5°F

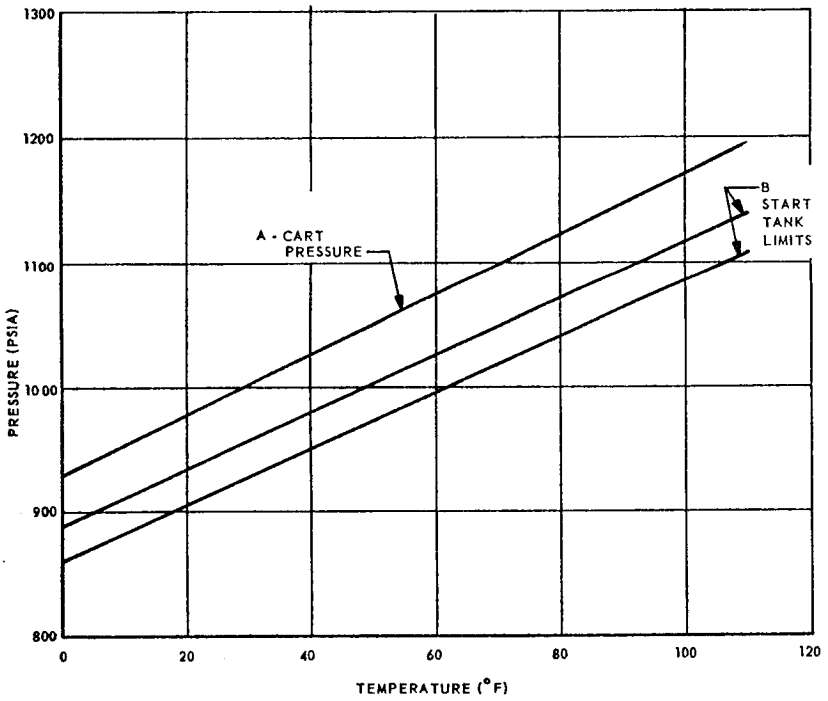


Fig. 3-5f Oxidizer To Be Loaded at 90° ±5°F

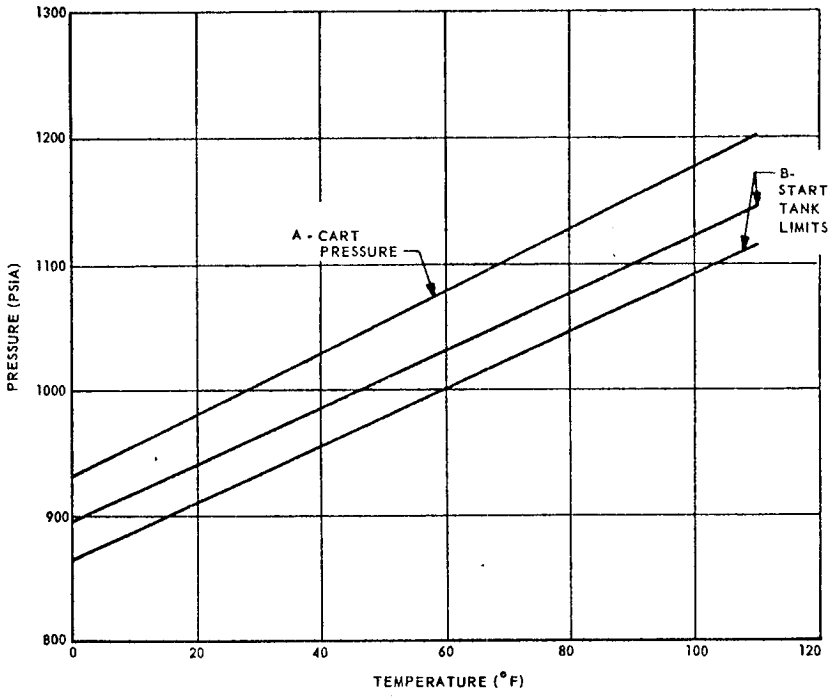


Fig. 3-5g Oxidizer To Be Loaded at 100° ±5°F

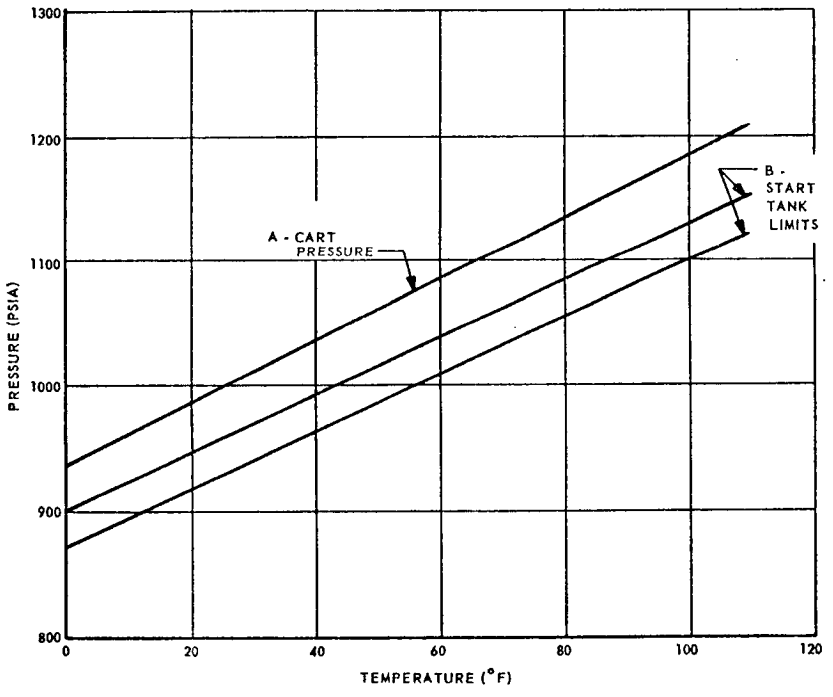


Fig. 3-5h Oxidizer To Be Loaded at 110° ±5°F

Section 4
ENGINE OPERATION

When propellants are supplied to the turbopump inlets at suitable pressures, engine operation is initiated by applying a 28-vdc starting signal. However, during the coast period prior to engine start, the lack of gravitational force would permit the gaseous ullage to enter the propellant feed lines. If a system was not provided to prevent gas from entering the feed lines during engine starting (includes initial start and subsequent restarts), then interrupted burning and possible engine malfunction could occur.

4.1 ULLAGE ORIENTATION CONTROL

During coast, a vehicle in a relatively low orbit, such as the Gemini-Agena Target Vehicle (ATV), is in a "near zero, relative g" field. The centrifugal force of the vehicle rotating around the earth and the earth's gravitational force cancel out each other, leaving as the only unbalanced force the slight deceleration (approximately 10^{-5}g) due to aerodynamic drag forces on the vehicle. Since the Gemini ATV is a "forward flying" satellite, the propellants seek equilibrium positions in the forward ends of the respective tanks.

After a booster-engine cutoff or an Agena main-engine shutdown, the residual propellants seek to become oriented at the forward end of the propellant tanks. In a "zero g" field, wetting propellants tend to climb the tank walls, thus increasing the liquid-gas interface and reducing the unwetted area. For small ullages or large amounts of fluid, the liquid may cover the walls completely and displace the ullage to the center of the propellant tank, thus indicating ullage orientation prior to Agena first-burn may not be required. In order to ensure Primary Propulsion System (PPS) start or

restart, propellant must fill the engine and feed lines and the containment "sumps" and cover tank outlets to a height sufficient to ensure that the pumps do not suck a hole through the liquid and ingest a prohibitive amount of pressurization gas.

The containment system, described in Paragraph 2.3.2, is designed to trap a sufficient quantity of propellants over tank outlets, thus allowing start of and maintaining combustion in the main engine until engine thrust has oriented the bulk propellants over the tank outlets. However, because of multiple burn requirements of the ATV mission, Agena ullage orientation is to be accomplished by an artificial gravity field produced by the 16 lb-thrust units of the Secondary Propulsion System (SPS).

A study has been made to determine the thrust-time relationship required to properly orient propellants for engine start at thrust levels between 16 and 400 lb, and for vehicle masses between 2,000 and 16,000 lb. Results of this study show that for a given vehicle weight, the propellant orientation period is approximately inversely proportional to the square root of the applied thrust. Figure 4-1 is used for determining the length of the propellant orientation period (defined as the time from SPS ullage ignition to the time when a propellant head above pump inlets is obtained which is sufficient to ensure reliable main engine start). For the preliminary mission profile, with assumed propellant-orientation burn times of 74 and 23 sec for the 16 and 200 lb-thrust units, respectively, the 16 lb-thrust unit leaves approximately 63,400 lb/sec of impulse propellant for orbital rendezvous maneuvers; while the 200 lb-thrust unit leaves approximately 15,600 lb/sec of impulse propellant. (Note: values are approximate and are presented to illustrate relative quantities.)

Two ullage orientation periods are provided in which the 16 lb-thrust units are used. Each period is of sufficient duration to provide propellant orientation with only one unit functioning at the lower thrust limit (14.4 lb). The short

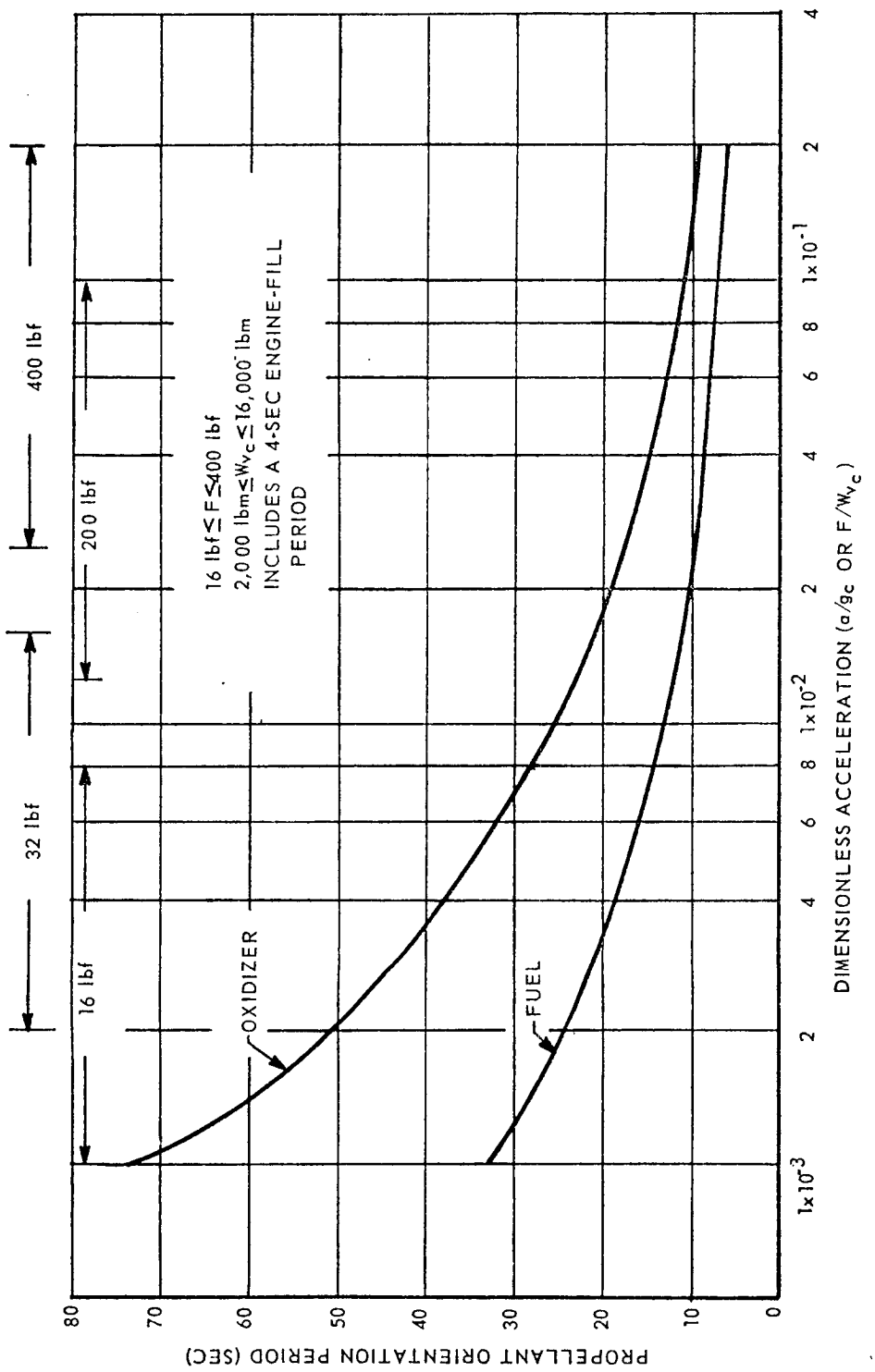


Fig. 4-1 Total Propellant Orientation Period Prior to Main Engine Ignition Versus Dimensionless Vehicle Acceleration

period (18 sec) is used prior to first burn when thrust/weight is the lowest. As this ratio increases, the ullage orientation period increases (Fig. 4-1). A 70-sec ullage orientation period is provided for use on all subsequent burns.

4.1.1 Secondary Propulsion System Operation

A brief discussion of the secondary propulsion system is included here, because of the use of the SPS in providing PPS ullage orientation control. Detailed SPS analysis is presented in the Secondary Propulsion System Engineering Analysis Report, LMSC-A387649-1.

The BAC Model 8250 secondary propulsion system (Fig. 4-2) is a storable liquid-propellant system which is designed to supply thrust, on demand, for multiple firings under vacuum and zero-gravity conditions for a period of 30 days. This system is completely contained, including positive expulsion propellant tanks, gas pressurization system, 16 lb-thrust (Unit I) thrust-chamber assembly, and 200 lb-thrust (Unit II) thrust-chamber assembly. A module consists of the components described above, and each vehicle has two modules – one located on the +Y axis and the other on the -Y axis. Each module is capable of producing a minimum total impulse of 40,000 lb-sec.

Components – The SPS fluid flow system (Fig. 4-3) includes the following components*:

(1) Spherical gas-storage tank providing high-pressure N_2-gas initially at a pressure of 4000 psig to the gas pressurization system for expelling the propellants and actuating the propellant valves.

(2) "On-off" type solenoid-actuated start valve isolating the high-pressure gas from the remainder of the pressurization system. An electrical impulse actuates the valve to pressurize the gas pressurization system.

* Paragraph numbers correspond to numbers on Fig. 4-3.

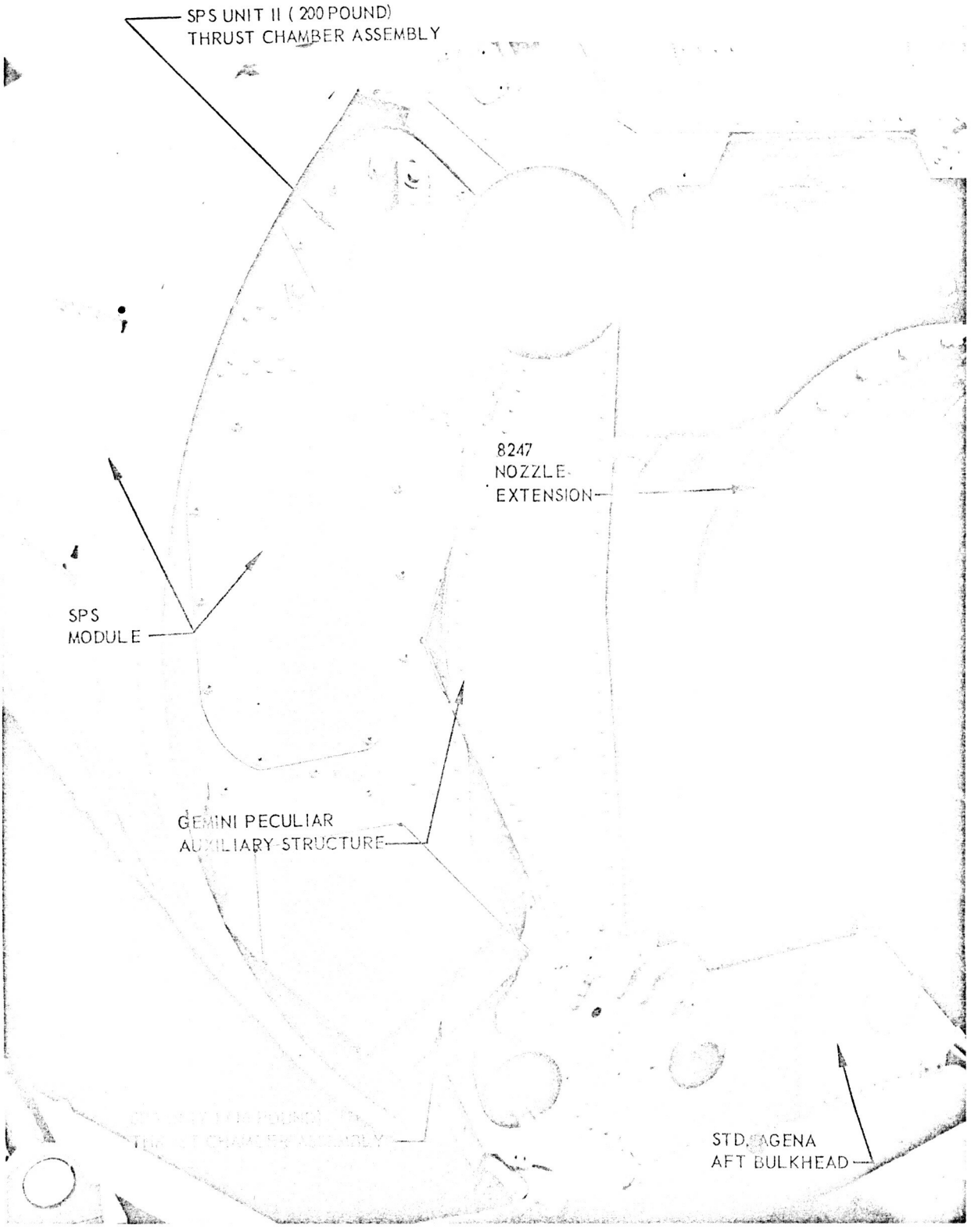


Fig. 4-2 Secondary Propulsion Module – Aft View

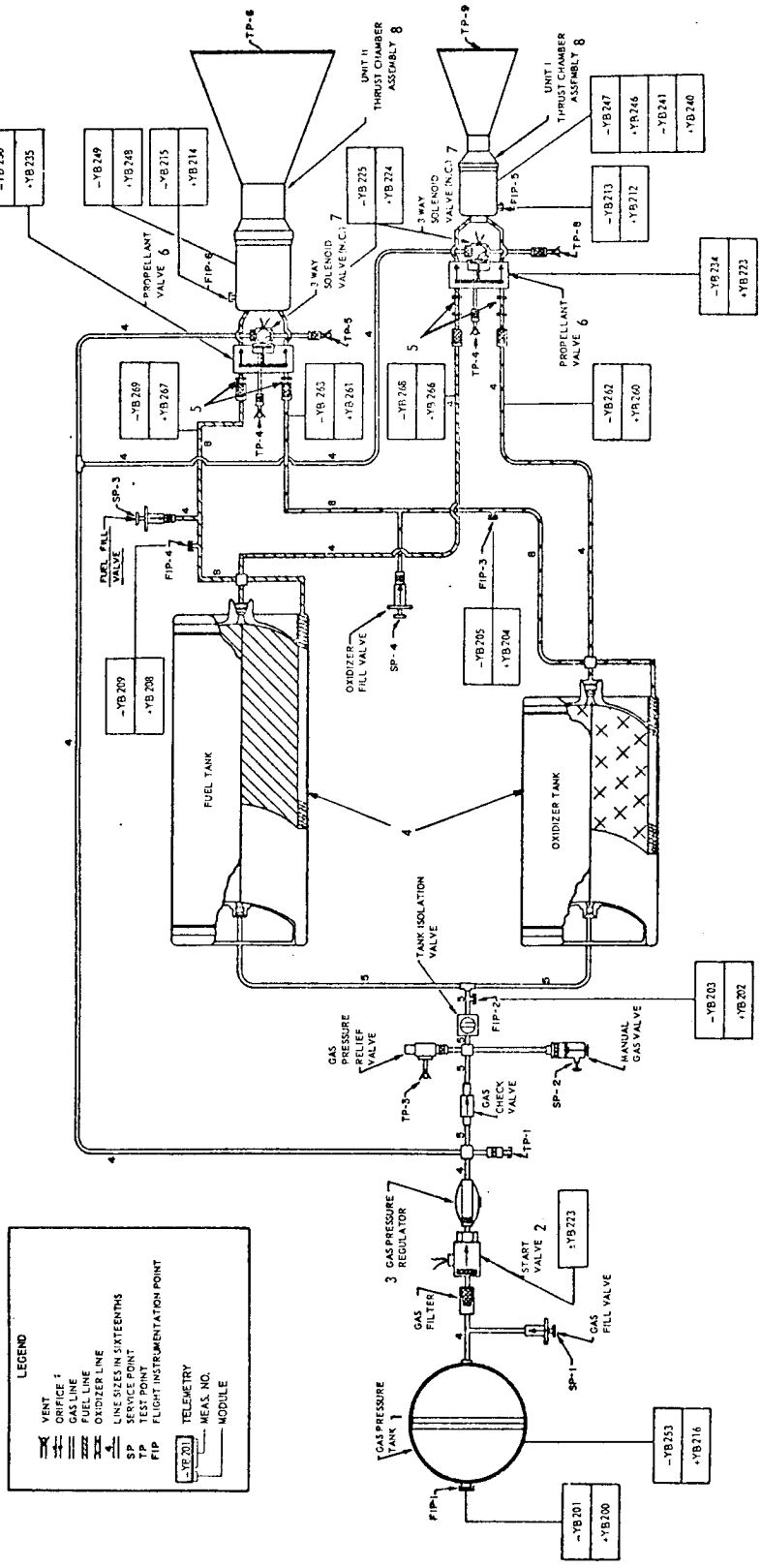


Fig. 4-3 Secondary Propulseion System Fluid Flow Schematic

(3) Single-stage gas pressure regulator for reducing the source gas pressure to an operating pressure of 205 psia for the entire pressurized-tank propellant-feed system and for propellant valve actuation.

(4) Cylindrical fuel and oxidizer tanks each containing a bellows-type metallic, multicycle, expulsion device which provides positive expulsion under zero-gravity conditions. Propellants are discharged from the tanks into the propellant feed lines to the Unit I and Unit II thrust-chamber assemblies.

(5) Trim orifices, located in the inlet of the thrust-chamber assembly propellant valve, balancing the propellant mixture ratio and providing for thrust-chamber interchangeability.

(6) "On-off" type solenoid-controlled gas-actuated propellant valves providing for sequencing of the fuel and oxidizer to the thrust-chamber assemblies.

(7) Three-way solenoid valves venting the propellant valve actuation pressure at shutdown and providing "on-off" control of the gas flow utilized for opening and closing the propellant valves.

(8) Unit I and Unit II thrust-chamber assemblies for producing thrust by the combustion of UDMH (fuel) and MON (oxidizer) at mixture ratios of 1.10 and 1.15 and chamber pressures of 79.0 and 96.0 psia, respectively. Starting is effected by the hypergolic ignition of fuel and oxidizer.

Operation – The secondary propulsion system is prepared for operation by electrically energizing the start valve 16 sec before operation of the propellant valves. A 16-sec start valve lead time assures that the propellant feed pressure is not below operating range before thrust-chamber operation starts. Accordingly, gas is permitted to flow from the gas pressure tank through the gas filter, start valve, gas pressure regulator, and gas check

valve into the gas side of the propellant tanks. The actuation-gas supply lines of the propellant valves are also pressurized. As the pressure downstream of the regulator increases to 227 psia, the regulator approaches lock-up.

After the pressurization period, the thrust-chamber assembly is operated by electrically energizing the solenoid propellant valve. Opening the valve establishes continuous liquid-propellant flow from the two propellant tanks through the propellant valve into the chosen thrust chamber. The gas pressure regulator maintains an automatically controlled gas pressure on the bellows in the fuel and oxidizer to provide positive expulsion of the propellant at any altitude and zero-gravity conditions. Liquid propellants enter the thrust chamber with a very slight MON lead and ignite hypergolically. As the flow continues in a steady state with the gas pressure regulator maintaining rated pressure, the thrust chamber develops rated thrust. The operation cycle is terminated by simultaneously de-energizing the three-way solenoid valve and the start valve.

The SPS complements the Primary Propulsion System by providing thrust required to make orbit adjustments, when the required impulse is less than that which can be supplied by the PPS, in addition to providing ullage orientation control. Unit I and Unit II thrust chambers are always operated in pairs for balanced thrust. The Unit II thrust chambers are used for orbit adjust, and the Unit I thrust chambers are used for ullage orientation.

4.2 ENGINE START SEQUENCE (FLIGHT)

When propellants are available at the tank outlets at sufficient head to prevent gas ingestion, the engine is ready for operation. No electrically armed condition is required for the Model 8247 engine.

For the following discussion, refer to the propellant flow schematic (Fig. 2-2), and to the wiring diagram (Fig. 2-5).

At the predetermined time during the ullage orientation period, the engine start sequence begins with the application of a 28-vdc signal to pins A and B of connector J6000, along with a simultaneous grounding of pins D and E. The start signal passes through the electronic gate circuitry, which acts as a junction box for dispersing the signal to plug P6011. Pins H and Z of P6011 furnish power to the gas generator fuel solenoid valve (GGFSV); pins J and a furnish power to the gas generator oxidizer solenoid valve (GGOSV); and pins K and b supply power to the pilot-operated solenoid valve (POSV). The three solenoid valves actuate simultaneously upon receiving the start signal.

Prior to launch, the start tank bellows and lines are loaded with the proper volume of propellants. Opening of gas generator (GG) solenoid valves initiates start tank operation. The start tank ullages (outside of bellows) are charged to the prescribed pressure (approximately 1000 psia). With the opening of the solenoid valves, propellants flow into the gas generator due to the force of the gas pressure on the bellows. The propellants ignite hypergolically in the gas generator which provides pressure buildup and sufficient energy to rapidly overcome the turbine-wheel break-out torque, resulting in turbine acceleration. As propellant flow to the gas generator continues (due to bellows expulsion), the ullage volume in the start tanks increases, the pressure decays, and the turbine accelerates

rapidly simultaneously with propellant combustion in the gas generator. Since the pumps are coupled to the turbine wheel, the pump-outlet pressures rise consistent with the relatively high horsepower availability. The process of start tank pressure decay and pump outlet-pressure increase continues until the two pressures equalize, at which time propellant expulsion from the start tanks stops, because the pump-outlet pressure is sufficient to force propellant back into the tank, thereby compressing the bellows and starting the recharging process.

At this time, the gas generator is being fed with the propellants from the pumps and the bootstrap operation has been initiated. With the build-up of turbine speed and increase of pump-outlet pressure, the recharging process is maintained until the fuel and oxidizer valves open and thrust-chamber ignition occurs. After ignition, steady-state operation is achieved, and the start tank propellant supply is replenished and maintained by steady pump-discharge pressure. The check valves prevent backflow of propellants during orbital coast, thus continuing the flow of propellants within the start tanks and assuring subsequent starts.

At the same time that the GG solenoid valves open, the pilot-operated solenoid valve opens for initiating the sequence which shuts off flow through the fuel valve, thus allowing actuation pressure to build up as described in Paragraph 2.2.8. When the actuation pressure reaches 365 psig, the fuel valve main-poppet opens. On initial or phase "A" start, fuel flow to the thrust chamber does not take place until the main-discharge pressure reaches 525 psig and ruptures the frangible disc. During subsequent starts, fuel flows into the thrust chamber immediately upon opening of the fuel valve.

Simultaneously with the rise in fuel pump-discharge pressure, the oxidizer pump-discharge pressure increases; at 225 to 300 psig, the oxidizer valve is opened when a spring-loaded poppet is unseated. As soon as this poppet

opens, the frangible disc ruptures, since the rupture-pressure is 180 psig. Oxidizer then flows through the thrust-chamber cooling passages to the oxidizer injector manifold. For all practical purposes, initial and subsequent start operations are identical, because the frangible disc ruptures almost immediately upon opening of the poppet. Inasmuch as the frangible disc is only used to prevent leakage of propellant during the booster phase of flight, no problem exists.

Under normal operating conditions, oxidizer enters the thrust chamber first. Then fuel enters through the injector with two jets of fuel impinging on each oxidizer jet. These propellants ignite hypergolically to initiate thrust-chamber operation. The time from application of start signal to the time when 70 percent P_c (thrust chamber pressure) is reached is considered the start transient, and is nominally 1 sec. Steady-state thrust is defined as the time from 70 percent P_c start to P_c decay on engine shutdown. If operating temperatures are below normal, the fuel may enter the thrust chamber before the oxidizer without appreciable change in the engine start transients.

Once steady-state thrust is achieved, the engine continues to operate nominally as long as the start signal is applied.

4.3 ENGINE SHUTDOWN (NORMAL FLIGHT)

Removal of the start signal from pins A and B and the ground from pins D and E initiates the engine shutdown sequence. Loss of electrical power allows the solenoid valves (GGFSV, GGOSV, and POSV) to close under spring and dynamic load.

Closure of the POSV permits fuel valve-actuation pressure to decay, as fuel passes through, rather than stagnating and holding the fuel valve open, with the resultant shutting off of fuel flow to the thrust chamber and termination of combustion within. The closure of the GGFSV and GGOSV shuts off flow of fuel and oxidizer, respectively, to the gas generator and terminates combustion in the GG. The lack of combustion products causes the turbine to decelerate, and accordingly, the fuel and oxidizer pump pressures decrease. Decrease of the oxidizer pressure allows the spring-loaded oxidizer valve to close.

4.3.1 Engine Shutdown (Turbine Overspeed)

If at any time during engine operation a turbine overspeed condition (29,500 ±500 rpm) occurs, the engine shuts down automatically by the electronic gate which senses the overspeed condition as described in Paragraph 2.2.1. The electronic gate then trips the relay contacts K-1 and K-2 (Fig. 2-6) to the open condition, thus interrupting the 28-vdc signal to the solenoid valves. Removal of electrical power allows the solenoid valves (GGFSV, GGOSV, and POSV) to close under spring load, thereby initiating the shutdown sequence previously described.

Reset coils are provided so that the relay contacts may be returned to the closed "flight ready" condition for enabling subsequent attempts at normal engine operation. Thus, one engine malfunction does not necessarily mean that a restart can not be attempted.

Turbine overspeed might be expected if propellant exhaustion occurs. Under this condition, there is no load on the turbopumps, and the propellants in the start tanks are free to empty into and sustain combustion in the gas generator, thus accelerating the turbine wheel and turbopump assembly until shutdown by the overspeed circuitry of the electronic gate.

4.4 ROCKET ENGINE GROUND OPERATION

The ground firing sequence is similar to that for flight operation, except for the use of additional ground safety features which are test facility hardware only and are removed for flight operation. The safety features (including necessary facility wiring and/or plumbing) are as follows:

(1) Gas generator manifold-pressure switch (GGMPS)
(2) Thrust-chamber pressure switch (TCPS)
(3) Propellant tank differential-pressure switch
(4) Emergency fuel, oxidizer, and helium-vent valves

All engine commands, during ground firing, are relayed through facility timers. If the TCPS and GGMPS do not "make" within the specified time, automatic termination of all electrical power to the engine is initiated at the facility. Should malfunction come prior to the first 143 sec of engine firing, the facility emergency relief ports automatically open.

Gas Generator Manifold-Pressure Switch – The GG manifold pressure switch is designed to remove the facility electrical power from the engine, if pressure within the gas generator does not reach 312 psig (nominal) within 0.5 sec after application of the start signal. This de-energizes the solenoid valves for shutting down the engine as described in Paragraph 4.3.

The GGMPS is required to prevent detonation of propellants that might have leaked into the GG manifold. During ground firings, when the vehicle is in a vertical position, the gas generator is in a horizontal position. If one of the gas generator solenoid valves leak, propellant could accumulate in the manifold and create a potentially explosive condition. This condition does not exist in flight, since the liquid is free to bleed to ambient (vacuum) through the turbine exhaust duct.

Thrust-Chamber Pressure Switch – The Thrust-Chamber Pressure Switch initiates the engine-shutdown sequence, if thrust-chamber pressure (P_c) does not reach 372 psig (nominally) within two sec after application of engine start signal, or at any time P_c falls below 372 psig (nominal) during normal burn.

Facility P Pressure Switch – The facility P pressure switch actuates facility emergency-vent valves if at any time the fuel tank pressure drops within 2.5 psid of oxidizer tank pressure with propellants in the tanks. This functions to preclude the possibility of excessive oxidizer tank pressure causing bulkhead reversal.

Facility Emergency-Vent Valves – Facility emergency-vent valves are provided for use in the event of malfunction of the helium sphere and propellant tanks. After first actuation of the pilot-operated helium-control valve, there is no provision on the vehicle for venting the pressurization system. Should it become necessary to stop the engine early in the test, the helium sphere would continue to blow down into the propellant tanks and would eventually rupture them. With the addition of the facility emergency-vent valves, any time that the engine is shut down during the first 143 sec of firing, the helium sphere and propellant tanks are automatically vented. The emergency vent valves are also actuated by the facility P switch, in the event that fuel tank pressure drops within 2.5 psid of oxidizer tank pressure.

The emergency vent valves are strictly a facility item. Vehicle plumbing has been altered to allow access to the helium sphere pressure and propellant tank pressure.

Section 5
INSTRUMENTATION

5.1 INTRODUCTION

Instrumentation requirements for the Gemini-Agena Target Vehicle (ATV) comprise three categories as follows:

- Prelaunch
- Flight telemetry
- In-flight status display

Prelaunch instrumentation monitors the status of various Propulsion System components to determine vehicle readiness for flight. Measurements taken during flight, and relayed to tracking stations by telemetry, are those which are used to determine the flight performance of the Primary Propulsion System (PPS).

Information presented on the Agena status panel (ASP) is necessary for the astronaut's safety and operation of the ATV.

5.2 PRELAUNCH INSTRUMENTATION REQUIREMENTS

5.2.1 Primary Propulsion System

Table 5-1 lists the prelaunch measurements which are monitored while the vehicle is on the pad. Countdown critical transducer readings are transmitted to the control center by the vehicle umbilical J-100 which is disconnected at launch.

Launch or Hold Criteria – The first group listed in Table 5-1 concerns those measurements which determine if the ATV is ready for launch.

Table 5-1

PRIMARY PROPULSION SYSTEM PRELAUNCH INSTRUMENTATION REQUIREMENTS (LANDLINE)

	Measurement	Transducer Range	Maximum	Minimum
1.	Oxidizer Start Tank Pressure, psi	0 to 1500	Ref. Fig. 3-5	
2.	Fuel Start Tank Pressure, psi	0 to 1500	Ref. Fig. 3-4	
3.	Oxidizer Start Tank Temperature Monitor, ^{o}F	-10 to +150	115	35
4.	Fuel Start Tank Temperature Monitor, ^{o}F	-10 to +150	115	35
5.	PPS Oxidizer Tank Temperature Monitor, ^{o}F	40 to 150	60	45
6.	PPS Fuel Tank Temperature Monitor, ^{o}F	40 to 150	60	45

Table 5-1 (Cont.)

Measurement	Transducer Range	Maximum	Minimum
7. PPS Oxidizer Tank Pressure Monitor, psig	0 to 80	32	30
8. PPS Fuel Tank Pressure Monitor, psig	0 to 80	40	38
9. Helium Gas Temperature, °F	40 to 150	165	—
10. Helium Supply Pressure Monitor, psig	0 to 3000	Ref. Fig. 5-1	
11. Oxidizer Pump Lip-Seal Pressure Monitor, psig	0 to 25	8	2

- Values Derived from Above Readings Determining Launch or Hold

1. Propellant Tank Differential Pressure			
Fuel over Oxidizer, psid	—	34.5	0
Oxidizer over Fuel:			
Empty Propellant Tanks, psid	0 to 34.5	—	—
Propellants in Tanks, psid	5 to 34.5	—	—

- Additional Prelaunch Instrumentation Requirements

1. N_2 High-Pressure and Lip-Seal Valve Open Monitor
2. Lip-Seal Valve Control (Open)
3. Propellant Isolation Valve Control (Open)
4. Propellant Isolation Valve (Position)

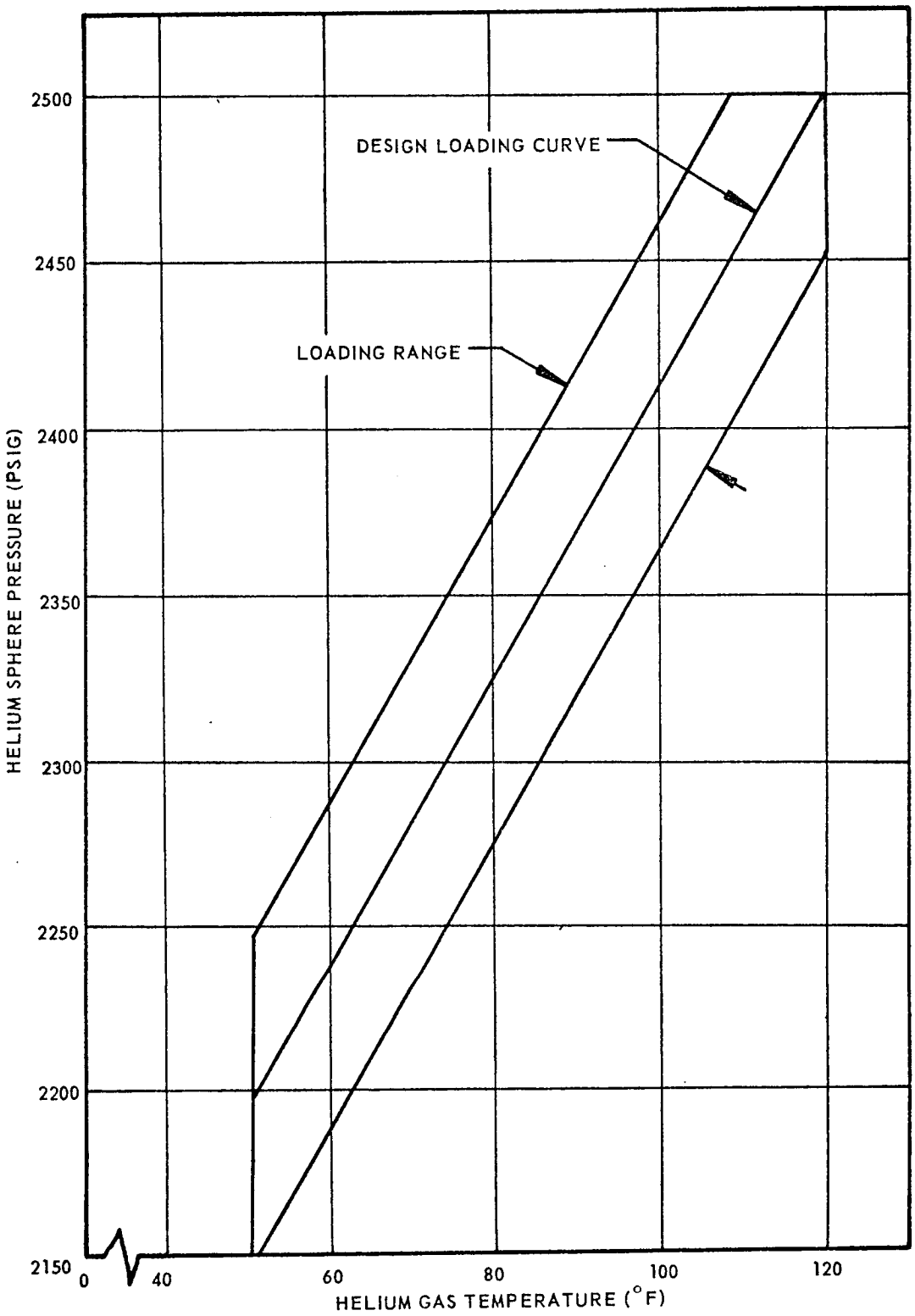


Fig. 5-1 Primary Propulsion System Prelaunch Helium Loading

Start Tanks – Start tank temperatures and pressures are used to determine start system readiness immediately prior to the vehicle launch. Transducers indicating start tank pressures are those which sense fuel and oxidizer venturi inlet pressures. Temperature-pressure relationships in the start tanks are established when the tanks are pressurized with nitrogen during loading. Any change in pressure, which does not relate to a corresponding change in temperature, may be sufficient reason for demating the Gemini-ATV from the booster and investigating the possibility of component leakage within the start system. System instrumentation errors average out and do not affect the readings, and "Go," "No-Go" decisions are based upon the change in pressure only. This pressure change, with no leakage, is slight and influenced by temperature only. Figures 3-4a through 3-4h show fuel start tank and Figs. 3-5a through 3-5h depict oxidizer start tank pressure-temperature relationships.

Main Propellant Tanks – Temperatures of the propellants in the main tanks are required to determine propellant specific gravity which varies with temperature. The "Go," "No-Go" launch criteria are based on the temperature range of both propellants (45° to 60°F), and the temperature differential between the two propellants (7°F max.). Since specific gravity of the propellants varies with temperature (Fig. 3-1) and the volume of propellants loaded is constant, the weight of impulse propellants is inversely proportional to temperature. Thus, if propellant temperature is too high, the mass of propellants loaded could be volumetrically insufficient to provide minimum burn time and accomplishment of mission objectives. For example: the weight difference between 100 ft^3 of oxidizer loaded at 50°F and 100 ft^3 loaded at 70°F would be approximately 125 lb, an equivalent of over 3 sec of main engine burn. A large temperature differential between the two propellants during engine firing would significantly alter the mixture ratio, because of density changes, and result in a lower than desired I_{sp}.

Propellant tank pressure monitors are required for the following reasons:

(1) To preclude the danger of propellant tank common bulkhead reversal

(2) To maintain proper liftoff ullage pressures

(3) To preclude the possibility of exceeding the safe operating pressures of the tanks

(4) To determine the pressure differential between fuel and oxidizer tanks, which varies as shown in Table 5-1.

Helium Sphere – Helium-sphere gas temperature and pressure are monitored, during loading, in order to accurately fill the sphere with the proper mass of gas (Fig. 5-1). This parameter is also required to limit pressure to a value commensurate with the structural strength of the sphere. At temperatures above 165°F, the helium sphere structural strength decreases and the pressure at which the sphere will burst commensurately decreases. Therefore, limiting the loading rate and keeping the temperature below this value are necessary.

Oxidizer-Pump Lip Seal – A pressure of 2 to 8 psig is required for lip-seal pressurization. This parameter is necessary to ascertain proper functioning of the lip pressurization system prior to launch, as a "Go," "No-Go" parameter.

Additional Prelaunch Instrumentation Requirements – To assure the presence of the oxidizer-pump lip-seal purge during propellant loading prior to launch, the position of the lip-seal valve must be known and the means of controlling the valve position must be provided.

Similarly, knowledge of the position and the capability of propellant isolation valves (PIV's) are necessary for the proper loading of the main propellant tanks. The PIV's must be in the open position for propellant loading and ascertained open immediately prior to liftoff.

5.2.2 Secondary Propulsion System

Table 5-2 lists Secondary Propulsion System (SPS) prelaunch instrumentation requirements and presents the qualifying range for launch. These parameters, when within this required range, indicate that the SPS has been loaded properly and should function normally in flight.

Table 5-2

SECONDARY PROPULSION SYSTEM PRELAUNCH INSTRUMENTATION REQUIREMENTS (LANDLINE)

	Measurement	Transducer Range	Maximum	Minimum
1.	Nitrogen Gas Sphere (+Y), °F	0 to 150	100	0
2.	Nitrogen Gas Oxidizer Tank (+Y), °F	0 to 150	100	0
3.	Nitrogen Gas Fuel Tank (+Y), °F	0 to 150	100	0
4.	Nitrogen Gas Fuel Tank (-Y), °F	0 to 150	100	0
5.	Nitrogen Gas Manifold (+Y), psig	0 to 300	210	190
6.	Nitrogen Gas Manifold (-Y), psig	0 to 300	210	190
7.	Nitrogen Gas Sphere (+Y), °F	0 to 150	100	0
8.	Nitrogen Gas Oxidizer Tank (-Y), °F	0 to 150	100	0
9.	Nitrogen Gas Sphere (-Y), °F	0 to 150	100	0
10.	Nitrogen Gas Sphere (-Y), psig	0 to 4500	See Fig. 5-2	
11.	Nitrogen Gas Sphere (+Y), psig	0 to 4500	See Fig. 5-2	

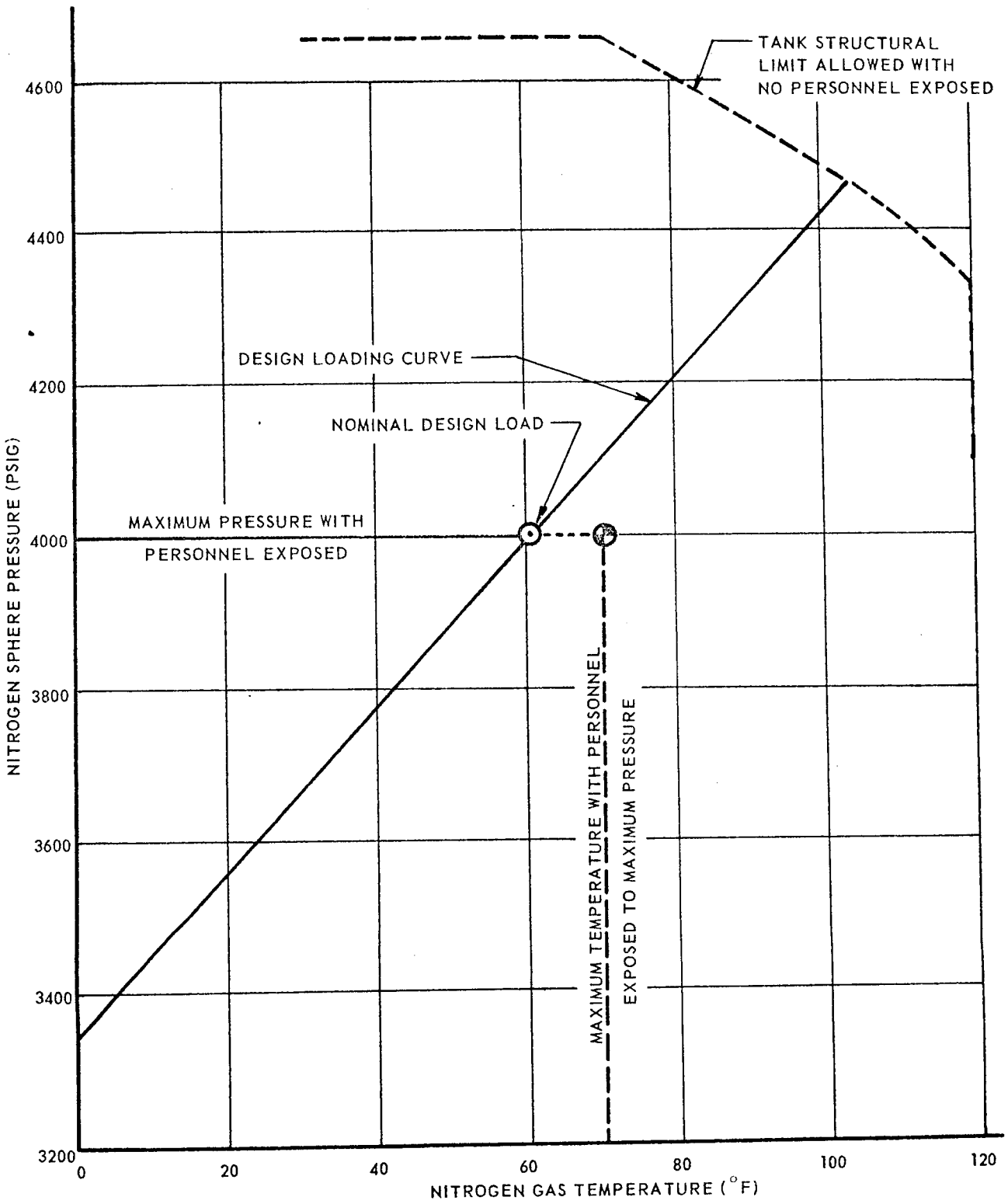


Fig. 5-2 Secondary Propulsion System Prelaunch Nitrogen Loading

5.3 FLIGHT INSTRUMENTATION – TELEMETRY

5.3.1 Primary Propulsion System Instrumentation

Table 5-3 lists the measurements necessary for satisfactorily determining the flight performance of the Gemini-ATV Primary Propulsion System. Information from the transducers is relayed to ground tracking stations by PCM telemetry. At the stations, the data are recorded by tape recorders and then are played back and printed on paper tapes. Transducer locations are shown schematically in Fig. 2-2.

Table 5-3
PRIMARY PROPULSION SYSTEM FLIGHT INSTRUMENTATION

Measurement No.	Sample Rate (per sec)	Measurement	Transducer Range
*B-1	16	Fuel Pump Inlet Pressure, psig	0 to 100
*B-2	16	Oxidizer Pump Inlet Pressure, psig	0 to 100
B-3	16	Turbine Manifold Pressure No. 1, psig	0 to 750
B-6	96	Combustion Chamber Pressure No. 1, psig	0 to 750
B-7	16	Helium Supply Pressure, psia	0 to 4000
B-8	16	Oxidizer Tank Pressure, psig	0 to 60
B-9	16	Fuel Tank Pressure, psig	0 to 60
B-11	16	Oxidizer Venturi Inlet Pressure, psig	0 to 1500
B-12	16	Fuel Venturi Inlet Pressure, psig	0 to 1500
*B-139	32	Model 8247 Engine Switch Group, v	0 to 5

*Standard Agena Measurement

Table 5-3 (Cont.)

Measurement No.	Sample Rate (per sec)	Measurement	Transducer Range
*B-31	1	Fuel Pump Inlet Temperature, °F	0 to 100
*B-32	1	Oxidizer Pump Inlet Temperature, °F	0 to 100
*B-35	DD	Turbine Speed	LSB**
B-68	1	Helium Sphere Temperature, °F	0 to 120
B-71	1	Oxidizer Pump Lip-Seal Pressure, psia	0 to 30
B-82	32	Fuel Valve Actuation Pressure, psig	0 to 1500
B-83	1	Thrust-Chamber Skin Temperature, °F	-50 to +150
*B-91	16	Combustion-Chamber Pressure No. 3, psig	475 to 550
B-96	1	Fuel Tank Temperature No. 1, °F	-15 to +170
B-97	1	Fuel Tank Temperature No. 2, °F	-15 to +170
B-130	32	Propellant Isolation Valves Open/Closed, v	0 to 5
B-132	32	Turbine Manifold Pressure No. 2, psia	0 to 120
B-136	1	Oxidizer Tank Skin Temperature (+Z), °F	-15 to +170
B-137	1	Oxidizer Tank Skin Temperature (-Z), °F	-15 to +170
B-141	1	Oxidizer Start Tank Temperature, °F	-10 to +150
B-142	1	Fuel Start Tank Temperature, °F	-10 to +150
B-148	32	Oxidizer Injector Pressure, psig	0 to 1000
B-184	1	Nozzle Extension Skin Temperature No. 1, °F	-200 to +800
B-185	1	Nozzle Extension Skin Temperature No. 2, °F	-200 to +800

*Standard Agena Measurement
**Least significant bit

Note from Table 5-3 that the Gemini-ATV instrumentation requirements are much more extensive than those for the standard Agena D (standard Agena D measurements prefixed with an asterisk). The reasons for these differences are that the Model 8247 is a new engine without previous flight history, and present mission profiles require that the ATV have an extended orbital period. Thus, more diagnostic measurements are necessary in order to ascertain the behavior of the propulsion system during the orbital period, the pre-dock period, orbital maneuvers, and the post-dock period.

Pump Inlet Pressures – Fuel and oxidizer pump inlet pressure transducers (B-1 and B-2, respectively) are basic Agena instrumentation needed for determining total propellant flow rate which is necessary to calculate Specific Impulse (I_{sp}). These transducers serve as backup instrumentation to tank top pressures (B-8 and B-9) but may not be used as a substitute for them. B-1 and B-2 measurements give total average inlet pressures (suction pressures) for the turbopumps during an engine burn, which is one of the factors in determining total flow rate. Pump inlet pressures are necessary in making calculations for the Subsystem Technical Analysis Report (STAR), which evaluates the propulsion system performance in flight.

Turbine Manifold Pressure No. 1 – Turbine manifold pressure No. 1 (B-3) is used to assess operation of the gas generator, turbine manifold, and turbine wheel.

Combustion-Chamber Pressure No. 1 – Combustion chamber pressure No. 1 (B-6) is used in determining thrust-chamber start and shutdown transients. This measurement provides information necessary for determining the start and shutdown impulse for each engine burn. The values obtained from B-6 are used only during engine start or shutdown; measurement B-91 (expanded scale) is used for steady-state thrust-chamber performance.

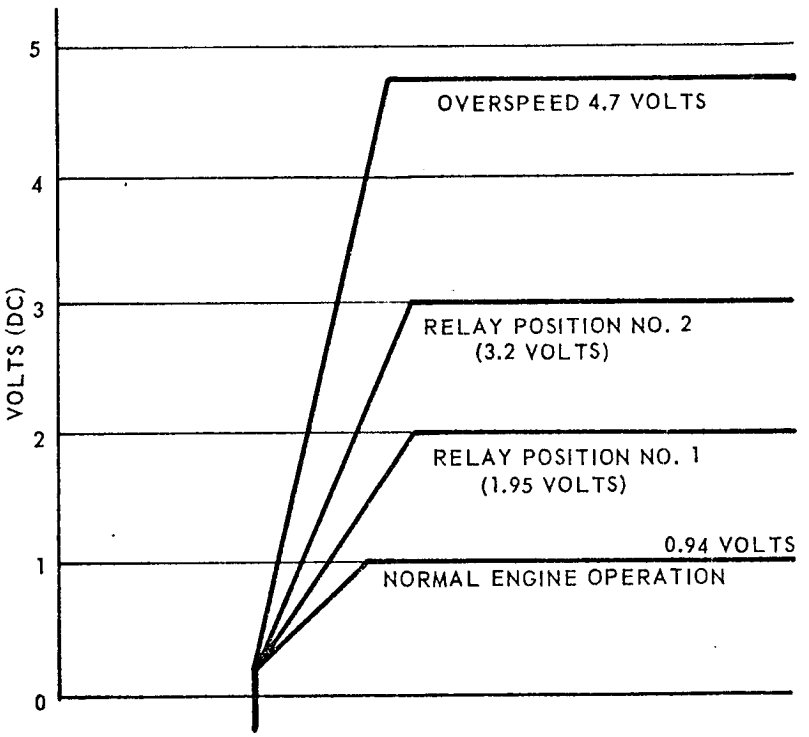
Helium Supply Pressure – The helium supply pressure (B-7) is monitored during first burn only; this measurement indicates proper operation of the pyrotechnically operated helium-control valve (POHCV). The pressure blowdown, as shown by B-7, will be compared to blow-down predictions run by computer at LMSC, Sunnyvale. This comparison will show total flight-pressurization system performance and indicate any abnormal areas. The measurement will also be utilized to determine any leakage of the high-pressure helium system during launch and prior to first burn.

This is the same transducer which is monitored on landline instrumentation during ground fill of the helium sphere.

Propellant Tank Pressures – Oxidizer and fuel ullage pressure measurements (B-8 and B-9, respectively) provide information on a number of parameters which are important to the success of the mission. The blow-down curves, generated by B-8 and B-9, will be assessed and compared to predictions (run during computer studies at LMSC, Sunnyvale) to ascertain that proper pressurization system operation has taken place during the blow-down phase of pressurization. During orbital flight, the B-8 and B-9 measurements will be monitored to determine any leakage in the propellant system upstream of the propellant isolation valves. Prior to any main engine firing, B-8 and B-9 will be checked to assure that sufficient pressure is present in the propellant tanks, to preclude the possibility of cavitation, upon start of the turbopumps. This parameter will be especially scrutinized prior to any main engine operation by the astronaut when he is in the vicinity of or docked to the Gemini-ATV. If leakage in the fuel system allows the oxidizer tank pressure to become greater than the fuel tank pressure, a bulkhead reversal could possibly take place resulting in damage and possible vehicle destruction. Monitoring, by ground stations, of B-8 and B-9 will yield information necessary to the astronaut's safety.

Propellant Venturi Inlet Pressure – Venturi inlet pressure transducers (B-11 and B-12) are basic instrumentation required for computing gas generator (GG) flow rates. The GG flow rates are needed to compute the total engine flow rate which is used in determining engine performance. These measurements will be the only means of determining pressure on the liquid side of the start tanks during the orbital coast period. Information from these transducers will be used to ascertain if the start system has been properly recharged during a previous engine burn. When an astronaut is docked to the vehicle, prior to his primary propulsion system firing initiation, this measurement must be determined to be within limits for a safe PPS start. These are the same transducers which are monitored on landline for start tank status prior to liftoff.

Model 8247 Engine Switch Group – The Model 8247 switch group (B-139), more accurately called engine "tell-tales," provides the only means of determining whether or not a turbine overspeed relay trip has occurred. This switch group has four voltage levels (Fig. 5-3) indicating the operational status of the relay contact matrix contained within the engine-mounted electronic gate. Prior to engine operation, the voltage level is zero. Upon application of electrical power to the engine, with normal engine operation, the level rises to 0.94v (nominal) and remains there throughout normal engine operation. In the event of turbine overspeed, relays K-1 and K-2 trip and the voltage level rises to 4.7v (nominal). Should one of the relays fail to open on overspeed shutdown, or fail to close on reset, this would be indicated by voltage levels for relay position Nos. 1 and 2. Relay position No. 1, at a nominal voltage level of 1.95, indicates that relay K_2 is open and K_1 is closed. If K_2 is closed and K_1 is open, this will be indicated by a voltage level of 3.2 (nominal) as the second relay position.



THESE VALUES ARE BASED ON 28V DC ON THE VEHICLE BUSS AND WILL VARY AND BE CALIBRATED FROM VEHICLE TO VEHICLE.

Fig. 5-3 Engine Switch Group Nominal Voltage Levels

Propellant-Pump Inlet Temperature – Fuel and oxidizer pump inlet temperatures (B-31 and B-32, respectively) are standard Agena-D instrumentation necessary for determining specific gravity of the propellants, which, in turn, is used in calculating propellant flow rate. Monitoring these parameters during coast periods following an engine burn will indicate the effect of heat transfer from the gearbox and turbine assembly. This is a diagnostic-type measurement to help determine the effects of PIV's and the venting of propellants overboard following an engine burn.

Turbine Speed – The turbine speed measurement (B-35) is basic instrumentation necessary for determining total fuel and oxidizer flow rates which are needed to determine engine performance. This parameter would indicate actual turbine speed in the event of a turbine overspeed condition. For diagnostic reasons, this measurement will show the start and shutdown transients. These data may be compared with PTVA data generated at Santa Cruz Test Base or Research and Development data generated at BAC to ascertain normal performance.

Helium Sphere Temperature – The helium sphere temperature measurement (B-68) is required for calculating the sphere polytropic blow-down exponent "N". Since "N" is used in predicting the blow-down characteristics of the sphere, this measurement is necessary for determining flight pressure-time histories which are compared with those assumed for the preflight calculations. This measurement is monitored on landline prior to launch.

Oxidizer-Pump Lip-Seal Pressure – This measurement (B-71) is required to assure that the required 2.0 to 8.0 psig oxidizer lip-seal pressure is maintained any time propellants are present in the engine.

Fuel Valve Actuation Pressure – The fuel valve actuation pressure (B-82) will indicate when the fuel valve opens during start transients. This measurment is required due to the Model 8247 engine starting sequence (fuel lead) being different than that of the Model 8096 engine. To determine nominal engine performance, this parameter will be reviewed.

Thrust-Chamber Skin Temperature – This measurement (B-83) is required for diagnostic reasons to assure that thrust-chamber skin temperature does not exceed the maximum permissible.

Combustion-Chamber Pressure No. 3 – The suppressed zero cumbustion-chamber pressure No. 3 (B-91) is basic instrumentation which is necessary for calculating total engine thrust. This is one of the required inputs for the STAR.

Fuel Tank Skin Temperatures – The two main fuel tank skin-temperature pickups (B-96 and B-97) are located on the +Z and -Z axes, respectively. These measurements are required to show the effect of exposure to space temperature and/or exposure to sun on vehicle propellants and to determine the vehicle absorptivity/emmissivity coefficient (α/ϵ). These measurements are Gemini-peculiar due to the extended orbital period.

Propellant Isolation Valves Open/Closed – The position of the propellant isolation valves is indicated by measurement B-130. The PIV's will be opened 4 sec prior to, and closed 4 sec after, PPS burn. This parameter is diagnostic and required to determine proper valve operation.

Turbine Manifold Pressure No. 2 – This measurement (B-132) is diagnostic and is used to determine proper turbine and gas generator operation.

Oxidizer Tank Skin Temperature – The two main oxidizer tank skin-temperature transducers (B-136 and B-137) are mounted on the +Z and -Z axes, respectively. These measurements are needed to show the effect of exposure to space temperatures and/or exposure to sun on vehicle propellants and to determine the vehicle absorptivity/emmissivity coefficient (α/ϵ). These measurements are Gemini-peculiar due to the extended orbital period.

Propellant Start-Tank Temperatures – The oxidizer and fuel start tank temperatures (B-141 and B-142, respectively) are required to determine the mass of gas present within the start tank, and together with venturi inlet pressures will indicate whether leakage has taken place in either the gas or liquid side of the start tanks. These are redline parameters, required prior to liftoff, and are monitored on landline.

Oxidizer Injector Pressure – This parameter (B-148) will be used to determine opening time of the main oxidizer valve. The measurement will also reflect proper turbopump/thrust-chamber operation.

Nozzle-Extension Skin Temperature – The two nozzle-extension skin temperatures pickups (B-184 and B-185) are located on the +Z and the -Z axes, respectively. These measurements are needed to determine if the maximum permissible temperatures of the material (titanium) of the nozzle extension has been exceeded.

5.3.2 Secondary Propulsion System

Table 5-4 lists the measurements necessary for satisfactorily determining the flight performance of the Gemini-ATV Secondary Propulsion System. Information from the transducers is relayed to ground tracking stations by PCM telemetry. Transducer locations are shown schematically in Fig. 4-3.

Table 5-4

SECONDARY PROPULSION SYSTEM
FLIGHT INSTRUMENTATION

Measurement No.	Sample Rate (per sec)	Measurement	Transducer Range
B-200	16	Gas Sphere Pressure (+Y), psia	0 to 4500
B-201	16	Gas Sphere Pressure (-Y), psia	0 to 4500
B-202	16	Tank Manifold Pressure (+Y), psia	0 to 300
B-203	16	Tank Manifold Pressure (-Y), psia	0 to 300
B-204	16	Oxidizer Feed Pressure (+Y), psia	0 to 300
B-205	16	Oxidizer Feed Pressure (-Y), psia	0 to 300
B-208	16	Fuel Feed Pressure (+Y), psia	0 to 300
B-209	16	Fuel Feed Pressure (-Y), psia	0 to 300
B-212	32	Unit I Chamber Pressure (+Y), psia	0 to 100
B-213	32	Unit I Chamber Pressure (-Y), psia	0 to 100
B-214	32	Unit II Chamber Pressure (+Y), psia	0 to 120
B-215	32	Unit II Chamber Pressure (-Y), psia	0 to 120
B-216	1	Nitrogen Gas Sphere Temperature (+Y), °F	-50 to +150
B-223	16	Start Valve On/Off (+Y & -Y), v	0 to 5*
B-224	16	Units I and II Pilot Solenoid On/Off (+Y), v	0 to 5**
B-225	16	Units I and II, Pilot Solenoid On/Off (-Y), v	0 to 5**
B-233	1	Unit I Bipropellant Valve Temperature (+Y), °F	-50 to +250
B-234	1	Unit I Bipropellant Valve Temperature (-Y), °F	-50 to +250

*2.0 v = -Y on
3.0 v = +Y on

**2.0 v = Unit II on
3.0 v = Unit I on

Table 5-4 (Cont.)

Measurement No.	Sample Rate (per sec)	Measurement	Transducer Range
B-235	1	Unit II Bipropellant Valve Temperature (+Y), °F	-50 to +250
B-236	1	Unit II Bipropellant Valve Temperature (-Y), °F	-50 to +250
B-240	1	Unit I Injector Temperature (+Y), °F	-100 to +200
B-241	1	Unit I Injector Temperature (-Y), °F	-100 to +200
B-246	1	Unit I Skin Temperature (+Y), °F	32 to 2800
B-247	1	Unit I Skin Temperature (-Y), °F	32 to 2800
B-248	1	Unit II Skin Temperature (+Y), °F	32 to 2300
B-249	1	Unit II Skin Temperature (-Y), °F	32 to 2300
B-253	1	Nitrogen Gas Sphere Temperature (-Y), °F	-50 to +150
B-260	1	Unit I Oxidizer Feed-Line Temperature (+Y), °F	-50 to +150
B-261	1	Unit II Oxidizer Feed-Line Temperature (+Y), °F	-50 to +150
B-262	1	Unit I Oxidizer Feed-Line Temperature (-Y), °F	-50 to +150
B-263	1	Unit II Oxidizer Feed-Line Temperature (+Y), °F	-50 to +150
B-266	1	Unit I Fuel Feed-Line Temperature (+Y), °F	-50 to +150
B-267	1	Unit II Fuel Feed-Line Temperature (+Y), °F	-50 to +150
B-268	1	Unit I Fuel Feed-Line Temperature (-Y), °F	-50 to +150
B-269	1	Unit II Fuel Feed-Line Temperature (-Y), °F	-50 to +150

Secondary Propulsion System instrumentation is discussed in this report, because of the relationship of SPS to proper operation of the PPS.

Nitrogen Sphere Pressure – The +Y and -Y modules transducers (B-200 and B-201) are required to determine the mass of gas remaining during orbit for evaluating system capability. These transducers are also utilized to analyze and predict the condition of the Agena status panel (ASP), "SPS Hi" and "SPS Lo" lights. Nitrogen sphere pressure will be monitored continually for indications of high-pressure gas-system leakage.

Tank Manifold Pressure – The +Y and -Y modules measurements (B-202 and B-203) are required to verify energization of the start valve and gas pressure-regulator operation as follows:

(1) Provides verification that tanks are pressurized to operating pressure prior to thrust-chamber operation

(2) Analyzes condition of ASP lights

(3) Provides "second source" information for performance calculations (propellant flow rates)

(4) Is monitored continually during orbit to detect indications of low-pressure gas-system leakage.

Oxidizer Feed Pressure – The +Y and -Y modules transducers (B-204 and B-205) are required to calculate oxidizer flow rates, which are a function of pressure drop across the thrust-chamber assembly.

Fuel Feed Pressure – The +Y and -Y modules measurements (B-208 and B-209) are required to calculate fuel flow rates, which are a function of pressure drop across the thrust-chamber assembly.

Unit I Chamber Pressure – The +Y and -Y modules transducers (B-212 and B-213) provide the best single source of information for evaluating Unit I thrust-chamber assembly operation. Chamber pressures are required to calculate Unit I fuel and oxidizer flow rates, which are a function of pressure drop across the thrust chamber assembly. In addition, these measurements are required to calculate thrust and specific impulse.

Unit II Chamber Pressure – The +Y and -Y modules transducers (B-214 and B-215) provide the best single source information for evaluating Unit II thrust-chamber assembly operation. Chamber pressures are required to calculate Unit II fuel and oxidizer flow rates, which are a function of pressure drop across the thrust chamber assembly. In addition, these measurements are required to calculate thrust and specific impulse.

Nitrogen Sphere Temperature – The +Y and -Y modules measurements (B-216 and B-253) are required to determine mass of gas remaining during orbit for evaluating system capability. Information on SPS module orbital history is also provided by these measurements.

Start Valve ON/OFF (+Y and -Y) – Verification of proper system sequencing – i.e., "Ready" command – is provided by this parameter (B-223) which has a range of 0 to 5 volts. Applicable voltages are 2.0, confirming that the -Y start valve has been commanded to open, and 3.0, indicating that the +Y start valve has been commanded to open.

Units I and II Pilot Solenoid ON/OFF Module (+Y) – Verification of proper system sequencing – i.e., "Fire" command – is provided by this parameter (B-224) which has a range of 0 to 5 volts. Applicable voltages are 2.0, confirming that the Unit II solenoid has been commanded to energize, and 3.0, indicating that the Unit I solenoid has been commanded to energize.

Units I and II Pilot Solenoid ON/OFF Module (-Y) – Verification of proper system sequencing – i.e., "Fire" command – is provided by this parameter (B-225) which has a range of 0 to 5 volts. Applicable voltages are 2.0, confirming that the Unit II solenoid has been commanded to energize, and 3.0, indicating that the Unit I solenoid has been commanded to energize.

Unit I Bipropellant Valve Temperature – The +Y and -Y modules measurements (B-233 and B-234) are required to provide data (1) on probability of safe starts – i.e., propellant is not frozen, (2) for determining coast period requirements, and (3) on orbital thermal history of the SPS module.

Unit I Injector Temperature – The +Y and -Y modules measurements (B-240 and B-241) are required to provide data (1) on probability of safe starts at low temperatures (below $0^{\circ}F$) and (2) on orbital thermal history of the SPS module.

Unit I Chamber Skin Temperature – The +Y and -Y modules measurements (B-246 and B-247) provide data on the normality of Unit I thrust chamber operation.

Unit II Chamber Skin Temperature – The +Y and -Y modules measurements (B-248 and B-249) provide data on the normality of Unit I thrust chamber operation.

Unit I Oxidizer Feedline Temperature – The +Y and -Y modules measurements (B-260 and B-262) are required (1) to obtain specific gravity factors in Unit I oxidizer flow-rate determination, (2) to provide data on orbital thermal history of SPS modules, and (3) for use to predict performance degradation of Unit I thrust-chamber assembly, caused by oxidizer cavitation in the thrust-chamber trim orifices.

Unit II Oxidizer Feedline Temperature – The +Y and -Y module measurements (B-261 and B-263) are required (1) to obtain specific gravity factors in Unit II oxidizer flow-rate determination and (2) to provide data on orbital thermal history of the SPS modules.

Unit I Fuel Feedline Temperature – The +Y and the -Y modules measurements (B-266 and B-268) are required (1) to obtain specific gravity factors in Unit I fuel flow-rate determination and (2) to provide data on orbital thermal history of the SPS modules.

Unit II Fuel Feedline Temperature – The +Y and the -Y modules measurements (B-267 and B-269) are required (1) to obtain specific gravity factors in Unit I fuel flow-rate determination, and (2) to provide data on oribital thermal history of the SPS modules.

5.4 AGENA STATUS DISPLAY PANEL

The Gemini-ATV Status Panel (ASP), shown by Fig. 5-4, is mounted on the forward end of the Target Docking Adapter of the Agena Target Vehicle where it is visible to the astronauts in the Gemini spacecraft during and after the docking maneuver. The panel displays information on the status and safety of the Agena propulsion, guidance, electrical power, and docking systems. Originally, only eight Agena parameters were to be displayed in the Gemini spacecraft; however, the number of parameters increased to the point that the spacecraft no longer had the space or weight capability to accomodate them. Accordingly, the panel was placed on the Target Docking Adapter.

The ASP system consists of a display panel (Fig. 5-4) with nine display lights and three analog dials and the necessary circuitry which is distributed throughout the Gemini-ATV. When not in use, this system is normally de-energized in order to save power; however, the PPS and SPS Time Remaining Clocks are energized whenever the PPS or SPS engines fire.

Three of the twelve parameters displayed on the ASP panel, indicate PPS status and three indicate SPS status. The Primary Propulsion System displays are as follows:

- PPS Burn Time Remaining Clock
- "MAIN" Red Light
- "MAIN" Green Light

The Secondary Propulsion System displays are as follows:

- SPS Burn Time Remaining Clock
- "SEC HI" Green Light
- "SEC LO" Green Light

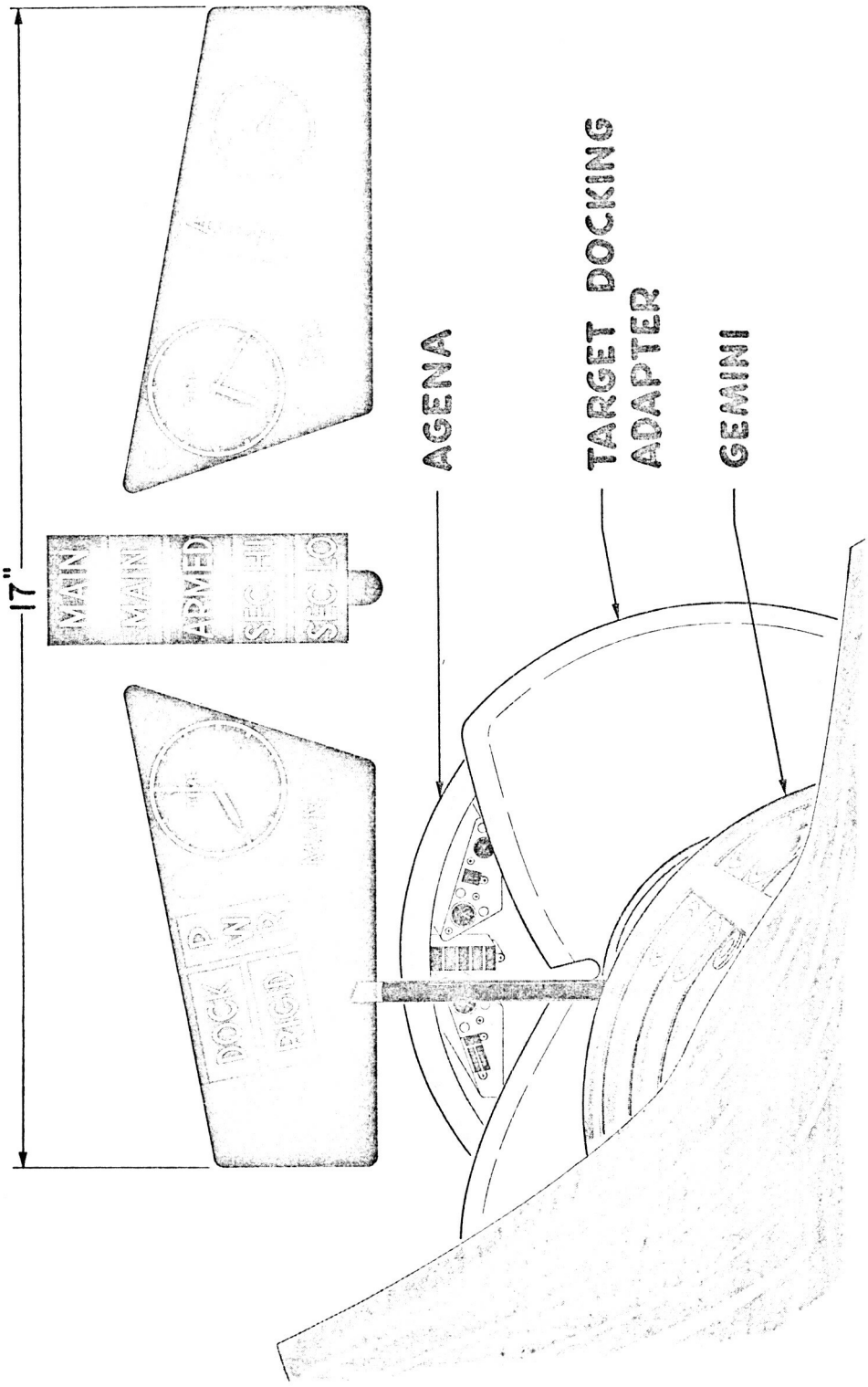


Fig. 5-4 Agena Status Display Panel

Requirements received from the astronauts stated that a dimming circuit must be added to the ASP system, because the panel lights are found to be too bright when the spacecraft docks with the ATV on the night side of the earth. Consequently, a dimming circuit has been added to reduce or increase the brightness to the desired level. The dimming circuit functions on all lights except MAIN Red which is always bright when "ON" for reasons outlined below. In order to improve reliability, two lamps are incorporated in each indicator light.

5.4.1 Time Remaining Clocks

Primary Propulsion System Clock – The Primary Propulsion System burn-time remaining display (Fig. 5-4) is a dc motor-driven clock with the short hand indicating the number of minutes (up to six) and the long hand the number of seconds (up to 60) of PPS burn-time remaining. The amount of time which the Primary Propulsion System will burn with the amount of fuel and oxidizer loaded have been previously calculated and the clock preset to this time via an AGE reset capability provided in the clock. This clock, with a decreasing time indication, runs whenever the PPS is burning and continually provides an indication of the amount of burn time left for operation of the system. The Primary Propulsion System clock is started by the "PPS Thrust Initiate" command, and stopped by the "PPS Thrust Cutoff" command.

Secondary Propulsion System Clock – The Secondary Propulsion System burn-time remaining display is a dc motor-driven clock similar to the one used for the PPS time display. The SPS clock can be run at two different speeds depending upon whether the Unit II high-thrust units (200 lb force) or the Unit I low-thrust units (16 lb force) are burning. The ratio used to differentiate between a Unit I and a Unit II firing is 12 to 1. For every second of Unit II firing, the clock registers 1 second less of time remaining, and for every 1 second of Unit I firing, the clock registers 1/12 of a second less of time remaining. This ratio of 12 to 1 was determined by taking a ratio of

the total mass-flow rate of propellants to the Unit II thrust chambers and dividing by the total mass-flow rate of propellants to the Unit I thrust chambers. The SPS Unit II thrust-time remaining is presented directly, whereas the clock reading must be multiplied by 12 to find the Unit I thrust-time remaining. The available equivalent burn-time for the SPS Unit II thrust chambers is preset into the clock via an AGE reset capability provided in the clock.

Originally the burn-time remaining analog displays were to be pulse-driven for accuracy; however, since these displays were a 5 to 10 percent indication, the high accuracy of a pulse-driven system was not necessary. Therefore, NASA directed that dc-driven clocks be used.

5.4.2 MAIN (RED) LIGHT

Light Operation – The MAIN (red) light (Fig. 5-5) may be turned "ON" by one of three ways: Turbine overspeed, hydraulic high-pressure-switch actuation, and fuel-oxidizer tank delta-pressure switch actuation.

Propellant Tank ΔP Switch – A differential pressure switch, mounted in the forward section of the vehicle and used to sense both fuel and oxidizer tank-top pressures, is set at 3 ±2 psid fuel-tank over oxidizer-tank. If the fuel tank pressure is less than 3 ±2 psid above the oxidizer tank, the only remaining prerequisite to light operation is the Gemini-ATV unregulated 28 vdc bus energization. Prior to launch and propellant tank prepressurizing, a blanket pressure (5 psi or less) is maintained on both propellant tanks equally; therefore, any time that the 28 vdc unregulated bus is on, the MAIN (red) light energizes. There is no cause for concern, since the light is extinguished when propellant tanks are pressurized.

Sufficient pressurization gas is loaded aboard the 3653-in.^{3} sphere to meet all mission requirements and maintain tank pressure differentials above the actuation value of the ΔP switch. Therefore, if the ΔP switch causes the MAIN (red) light to come on, the system has a leak.

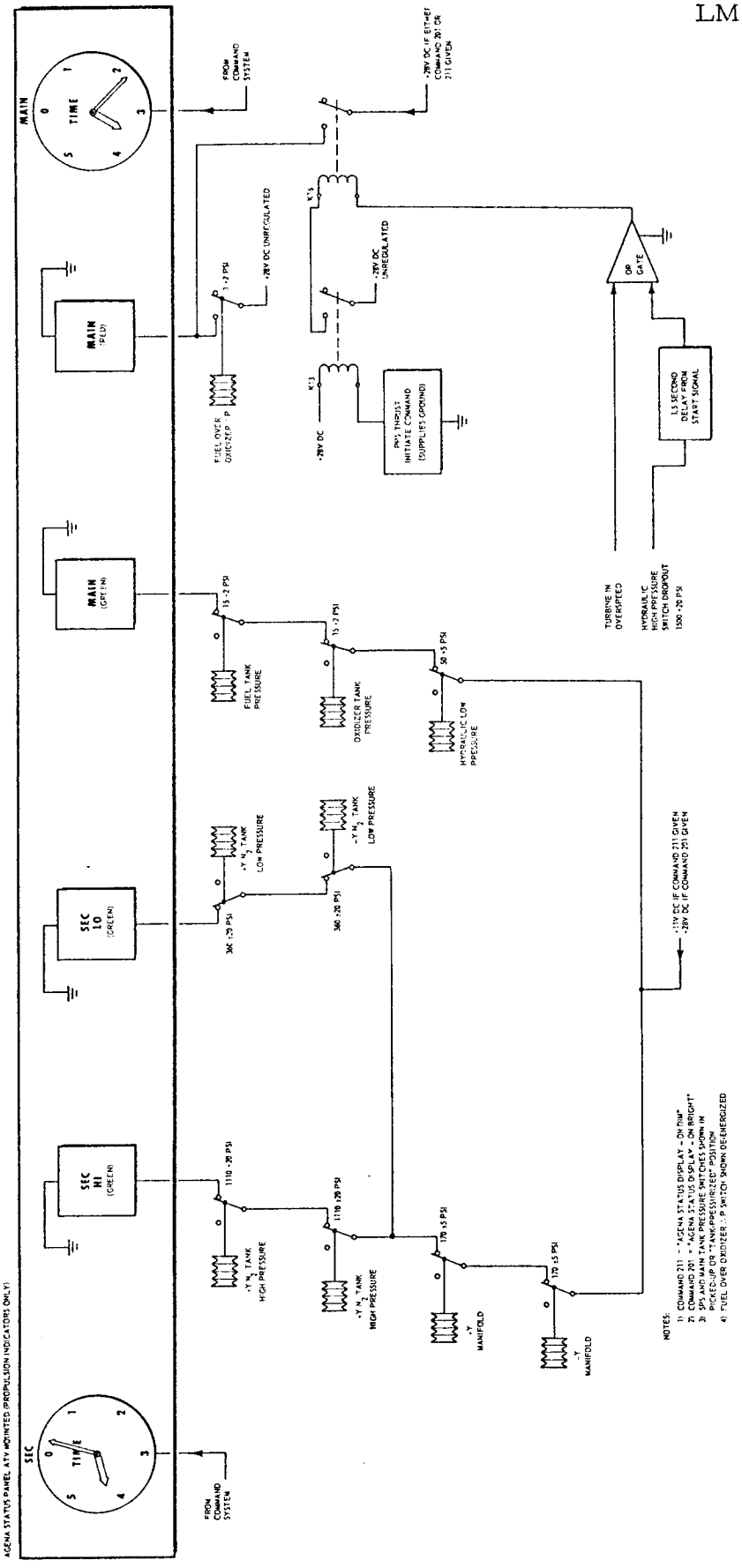


Fig. 5-5 Primary Propulsion System and Secondary Propulsion System Status Display Panel

A study of propellant time-pressure history was initiated and nominal curves, together with upper and lower 3-sigma limits, were developed (Fig. 6-3). Further studies were made with the aim of obtaining maximum warning time in the event of a pressurization system malfunction (leakage). An optimum oxidizer isolation valve closing-time was determined at 318 sec (Fig. 6-3) after initiation of the Model 8247 engine start signal. This time (318 sec) provides maximum capability for sensing imminent propellant tank bulkhead reversal (a possible catastrophic failure). The minimum pressure differential between tanks, 9 psid, occurs toward the last few seconds of PPS burn capability (using the lower 3-sigma fuel tank pressure and the upper 3-sigma oxidizer tank pressure). Utilizing the upper limit of the ΔP switch setting, the amount of fuel tank pressure leakage prior to ΔP switch actuation is 4 psi. At the opposite extreme, utilizing the upper 3-sigma limit of fuel tank pressure and the lower 3-sigma limit of the oxidizer tank pressure and taking the minimum allowed ΔP switch actuation point, a pressure leakage of 24 psi can occur before a switch actuation takes place. Therefore, if the ΔP switch actuates, a fuel tank pressure leakage of 4 to 24 psi will have occurred.

Hydraulic High-Pressure Switch – Referring to Fig. 5-5, note that when either command 201 or 211 is given, 28 vdc is supplied to normally open K-16 relay contacts. Once the "PPS Thrust Initiate" command is given, a ground return is supplied to relay coil K-13 contacts to the coil of relay K-16. Relay K-16 does not energize unless a ground path is created through the "OR" gate by either turbine overspeed or by the hydraulic high-pressure switch dropout. A 1.5 sec (measured from "PPS Thrust Initiate" command) delay is incorporated to allow time for the hydraulic pressure switch to build up to 1500 ±20 psi without igniting the MAIN (red) light. If this pressure switch fails to pick up during the allowed 1.5 sec or drops below switch setting anytime during normal engine operation, the circuit immediately supplies the required ground, through the "OR" gate, to relay K-16, closing K-16 normally open contacts, and thereby lighting MAIN (red) on the ASP.

Nominal hydraulic-discharge pressure is 2810 ±60 psig. The hydraulic pressure switch setting of 1500 psi was picked as a mean between engine controllability and indication of gross hydraulic system malfunction. Selection of the 1.5 sec delay to build up hydraulic-actuation pressure was based upon data generated during hydraulically controlled-engine testing indicating an approximate 1.0 sec delay maximum, over the specification extremes of temperature, to reach 1500-psi hydraulic-discharge pressure.

Turbine In Overspeed – Similar to the hydraulic high-pressure switch, the turbine in over-speed signal (Fig. 5-5) can trigger the "OR" gate and allow relay coil K-16 to energize by supplying a ground return at the "OR" gate, thus lighting the MAIN (red) light on the ASP. Nominal turbine speed is 24, 800 rpm. The signal to the "OR" gate is supplied by means of a frequency-sensitive relay system which triggers at 27, 000 rpm. This trigger value was obtained by analyzing engine test data and establishing a 3-sigma level below which a family of Model 8247 rocket-engine start transients would lie. If the turbine continues to accelerate, an automatic turbine overspeed shutdown is initiated by the engine-mounted electronic gate (see Paragraph 2.2.1 of this report) at 29, 500 ±500 rpm.

To determine rise times of the turbopump speed, a study was initiated with calculations based on assumed pump unloading factors. The critical shaft speed is 31, 000 rpm (no dwell allowed), and the rotor rotational stress lower-limit is 41, 000 rpm. The fastest rise time (approximately 40 ms) is calculated from 27, 000 (ASP "MAIN" red "ON") to 29, 500 rpm (automatic cutoff) and occurs with completely dry turbopumps. The slowest rise time (approximately 500 ms) is calculated from 27, 000 to 29, 500 rpm and occurs during a malfunction in which the fuel valve closes and the oxidizer continues to unload overboard. A most likely condition to cause turbine overspeed would be where maximum utilization of all PPS propellants was attempted and propellant exhaustion occurred, with oxidizer exhausting first and fuel exhausting soon afterward; a 12-lb fuel bias is included during propellant tank loading. (See Section 3 of this report). In the aforementioned case, the turbine speed

rise time is from 27,000 to 29,500 rpm in approximately 180 ms. Present Gemini-ATV mission profiles do not purposely include the intent to maximize usage of all propellants.

Reaction To MAIN Red During Flight – If the MAIN (red) light comes "ON" when the Primary Propulsion System is not firing, the malfunction indicated would be due to ΔP switch dropout and would indicate a possible tank bulkhead reversal. Should the light come "ON" during normal PPS burn, immediate remedial action is required by the spacecraft. Shutdown of the PPS must be initiated by issuing spacecraft command 500. If the MAIN (red) light goes "OFF" after command 500, the next ground station should have information, via telemetry, as to the cause of malfunction; i.e., whether the malfunction was due to turbine overspeed or hydraulic system malfunction. Should the light remain "ON", possible tank bulkhead reversal may take place, and the astronaut must take immediate appropriate action until the next ground tracking station can access the probable cause. Paragraph 5.2 of this report describes instrumentation available to determine possible failure modes.

Although the ASP has two light-intensity levels available, dim (command 211) or bright (command 201), the MAIN (red) light always illuminates bright when a malfunction occurs.

5.4.3 MAIN (Green) Light

Light Operation – The MAIN (green) light (Fig. 5-5) may be turned "OFF" by one of three ways: fuel tank pressure-switch dropout, oxidizer tank pressure-switch dropout, and hydraulic low-pressure-switch dropout. Figure 5-5 shows that either command 211 or 201 can supply power to the MAIN circuit and illuminate the MAIN (green) light, if the hydraulic pressure in the reservoir is above 50 psi, the oxidizer tank-top is above 15 psia, and the fuel tank-top is above 15 psia.

Hydraulic Low-Pressure Switch – The hydraulic low-pressure switch, set at 50 ±5 psia, is utilized to assure that proper hydraulic residual-pressure remains in the hydraulic reservoir. If this pressure should be lost, due to a leak or other reasons, the hydraulic pressure would not build up properly when the "PPS Thrust Initiate" command is sent. This could lead to instability of thrust direction during firing.

Oxidizer Tank Pressure Switch – The oxidizer tank pressure switch, set at 15 ±2 psia, is utilized to inform the astronaut if oxidizer tank pressure is decaying, due to a leak or other reasons. The turbopump requires a minimum head for proper operation and for suppressing cavitation. Oxidizer pump cavitation takes place below 10 psia at a propellant temperature of 60°F.

Fuel Tank Pressure Switch – The fuel tank pressure switch, set at 15 ±2 psia, is utilized to inform the astronaut if fuel tank pressure is decaying, due to a leak or other reasons. The turbopump requires a minimum head for proper operation and for suppressing cavitation. Fuel pump cavitation takes place below 13 psia at a propellant temperature of 60°F.

This switch is actually redundant with the ΔP switch located in the MAIN (red) circuit. If the MAIN (green) light is "ON", indicating sufficient oxidizer tank pressure to suppress cavitation; and the MAIN (red) light is not "ON", indicating the fuel tank pressure is at least 3 ±2 psi above the oxidizer tank, proper operation without the possibility of cavitation is assured.

Reaction to MAIN Green During Flight – The MAIN (green) circuit is designed to convey maximum information to the astronaut concerning the status of the Gemini-ATV and the permissibility of the Primary Propulsion System to fire. If the light is not "ON", no PPS firing should be attempted until the next ground station can, via telemetry, determine the cause. If the light should go "OFF" during PPS burn, cavitation of the turbopump may be imminent, and the engine may either be manually shut down, or the astronaut may rely on the engine-mounted turbine overspeed device to perform the shutdown.

5.4.4 SEC LO (GREEN) LIGHT

Light Operation – The SEC LO (green) light may be turned "OFF" by one or more of four ways: +Y nitrogen-gas tank low-pressure-switch dropout, -Y nitrogen-gas tank low-pressure-switch dropout, +Y manifold pressure-switch dropout, and -Y manifold pressure-switch dropout. Figure 5-5 shows that either command 211 or 201 will supply power to the SEC LO circuit and illuminate the SEC LO (green) light, if either the +Y or -Y nitrogen spheres are above 360 psia, or either the +Y or -Y manifolds are above 170 psia.

Manifold Pressure Switch – The nominal pressure experienced in the +Y or -Y manifold (identical modules) is 204 psia (nominal) regulated pressure. Allowed regulator deviations reduce this pressure to 197 psia (nominal), and the 3-sigma product variations further reduce the minimum manifold pressure levels to 190 psia (nominal). A value of 170 ±5 psia (dropout) for switch setting was finally calculated to allow 20 psi for temperature variations of the gas in the manifold while in orbit.

Nitrogen Gas Tank Low-Pressure Switches (+Y and -Y) – The +Y and -Y nitrogen spheres are each nominally loaded to 4000 psig. The low-pressure switches must operate in the aforementioned system environment but actuate (dropout) at 360 ±20 psia. Low-pressure switch setting is calculated on the basis that the minimum pressure would sustain one additional 150 sec (full duration) continuous Unit I operation. Due to the low propellant-flow rate, the final sphere pressure is 210 psia, allowing for proper regulator operation and, therefore, nominal thrust ratings throughout the firing duration.

Reaction To SEC LO (Green)Light During Flight – The SEC LO (green) light should be used in conjunction with the SPS clock as an indication of thrust-time available for Unit I thrust chambers. If the SEC LO (green) light goes "OFF" during an SPS Unit I normal burn sequence, the next ground station should be

consulted for information concerning remaining burn-time available for Unit I operation. Should the SEC LO (green) light go "OFF" when Secondary Propulsion System firing is not taking place, the loss of the light may be indicative of sphere or manifold leakage, and the next ground station should be consulted for the cause of light loss.

If the SEC LO (green) light is "OFF" and the SPS clock indicates time remaining in excess of one full-duration Unit I firing, a manifold leak may have developed. The manifold is the system most susceptible to leakage, since a small volume of gas is trapped between the start solenoid valve and tank bellows (Fig. 4-7). If a leak has developed, and the SEC LO (green) light goes "OFF" due to the pressure switch dropping below 170 psi, a start valve open command may be sent, and if SEC LO relights, a Unit I firing may be initiated.

5.4.5 SEC HI (Green) Light

Light Operation – The SEC HI (green) light may be turned "OFF" by one or more of four ways: +Y nitrogen-gas tank high-pressure-switch dropout, -Y nitrogen-gas tank high-pressure-switch dropout, +Y manifold pressure-switch dropout, and -Y manifold pressure-switch dropout. Figure 5-5 shows that either command 211 or 201 will supply power to the SEC HI circuit and illuminate the SEC HI (green) light, if either the +Y or -Y nitrogen spheres are above 1110 psia, or either the +Y or -Y manifolds are above the 170 psia.

The +Y and -Y manifold pressure switches are discussed in Paragraph 5.4.4, and as indicated by Fig. 5-5, these switches operate identically for the SEC LO and SEC HI light circuits.

Nitrogen Gas Tank High-Pressure Switches (+Y and -Y) – The +Y and -Y nitrogen spheres are each nominally loaded to 4000 psig. The high-pressure switches must operate in the aforementioned system environment but

actuate (dropout) at 1110 ±20 psia. High-pressure switch setting is calculated on the basis that the minimum pressure would sustain one additional 50 sec (full duration) continuous Unit II operation. For proper SPS regulator operation during a Unit II burn, the final sphere pressure must not drop below 400 psia for the relatively high propellant-flow rate.

Reaction To SEC HI (Green) Light During Flight – The SEC HI (green) light should be used in conjunction with the SPS clock as an indication of thrust-time available for Unit II thrust chambers. If the SEC HI (green) light goes "OFF" during an SPS Unit II normal burn sequence, the next ground station should be consulted for information concerning remaining burn-time available for Unit II operation. Should the SEC HI (green) light go "OFF" when Secondary Propulsion System firing is not taking place, the loss of the light may be indicative of sphere or manifold leakage, and the next ground station should be consulted for the cause of light loss.

If both SPS LO and SPS HI (green) lights go "OFF" simultaneously, this would be indicative of either +Y or -Y manifold leakage (manifold pressure-switch dropout). Should both lights go "OFF" and the Secondary Propulsion System is firing, then the SPS firing should be immediately terminated and the next ground station consulted for data indicating the cause. If both lights go "OFF" and the Secondary Propulsion System is not firing, then the SPS start valve open command (command 561) may be sent to open the start solenoid and repressurize the manifold to a value above 170 psi (manifold pressure-switch pickup pressure). Should the lights (HI and LO) remain "ON", an SPS firing may be attempted.

5.5 PRESSURE SWITCHES

Pressure switches are mounted on various PPS and SPS components to warn the astronaut when undesireable conditions develop within the propulsion systems. These switches operate at predetermined pressures and either extinguish a green light or light the MAIN (red) light, while maintaining a display to indicate the current status of the propulsion system.

5.5.1 Fuel Over Oxidizer ΔP

The fuel over oxidizer ΔP switch is a normally closed differential pressure-sensing switch mounted integrally on the main propellant tank assembly. This switch has high- and low-pressure inlets connected to the fuel and oxidizer tanks, respectively. The higher pressure on the fuel side holds the switch open. If fuel tank pressure drops to within 3 ±2 psid of oxidizer tank pressures, the switch returns to its normally closed position.

The switch employs twin mechanical bourdon tubes which are independant mechanically from each other. A design prerequisite was imposed upon the switch, such that any single malfunction would not allow a common passage between fuel and oxidizer systems. The switch has characteristics as shown in Table 5-5.

5.5.2 Fuel Tank Pressure Switch

The fuel tank pressure switch is a normally open pressure-actuated switch, connected to the ullage section of the fuel tank. This switch is connected in series electrically with the oxidizer tank pressure switch and the hydraulic low-pressure switch. Design characteristics are presented in Table 5-5.

Table 5-5

PRESSURE SWITCHES DESIGN CHARACTERISTICS

	Pressure
Fuel Tank Oxidizer Tank ΔP Switch	
Pressure, Range, psid	0 to 100
Proof Pressure (elements), psid	150
Burst Pressure (elements), psid	200
Line Pressure (maximum), psig	150
Burst Pressure (case), psig	700
Fuel Tank Pressure Switch	
Pressure Range, psia	0 to 100
Proof Pressure (element), psia	150
Burst Pressure (element), psia	200
Switch Point, psia	15 ±2
Oxidizer Tank Pressure Switch	
Pressure Range, psia	0 to 100
Proof Pressure, psia	150
Burst Pressure, psia	200
Switch Point, psia	15 ±2
Hydraulic Low-Pressure Switch	
Pressure Range, psia	0 to 100
Proof Pressure (element), psia	150
Burst Pressure (element), psia	200
Burst Pressure (case), psia	700
SPS Nitrogen Sphere Low-Pressure Switch	
Pressure Range, psia	0 to 4000
Burst Pressure (element), psia	8000
Burst Pressure (case), psia	4000

Table 5-5 (Cont.)

	Pressure
SPS Nitrogen Sphere High-Pressure Switch	
Pressure Range, psia	0 to 4000
Burst Pressure (element), psia	10,000
Burst Pressure (case), psia	4000
SPS Manifold Pressure Switch	
Pressure Range, psia	0 to 300
Burst Pressure (element), psia	600
Proof Pressure (element), psia	450
Burst Pressure (case), psia	700

5.5.3 Oxidizer Tank Pressure Switch

The oxidizer tank pressure switch is a normally open pressure-actuated switch, connected to the ullage pressure of the oxidizer tank. This switch is connected in series electrically with the fuel tank pressure switch and the hydraulic low-pressure switch. Design characteristics are shown in Table 5-5.

5.5.4 Hydraulic Low-Pressure Switch

The hydraulic low-pressure switch is a single-pole double-throw pressure-actuated switch set at 50 ±5 psia. Electrically, it is connected in series with the oxidizer and fuel tank pressure switches. Design characteristics are presented in Table 5-5.

5.5.5 Secondary Propulsion System Nitrogen Sphere Low-Pressure Switch

The SPS nitrogen sphere low-pressure switch is a single-pole single-throw pressure-actuated switch set at 360 ±20 psia. Design characteristics are shown in Table 5-5.

5.5.6 Secondary Propulsion System Nitrogen Sphere High-Pressure Switch

The SPS nitrogen sphere high-pressure switch is a single-pole single-throw pressure-actuated switch set at 1110 ±20 psia. Design characteristics are shown in Table 5-5.

5.5.7 Secondary Propulsion System Manifold Pressure Switch

The SPS manifold pressure switch is a single-pole double-throw pressure-actuated switch set at 170 ±5 psia. Design characteristics are shown in Table 5-5.

Section 6
PRIMARY PROPULSION SYSTEM PERFORMANCE AND DESIGN CHARACTERISTICS

6.1 INTRODUCTION

This section discusses the propulsion subsystem performance capability and the parameters governing engine performance and power level in vacuum flight. The method for predicting flight parameters and for determining flight performance are also presented.

6.2 ENGINE REQUIREMENTS

The pressurization and propellant feed systems supply propellants to the engine pump inlets at a sufficient pressure to permit the propellant pumps to operate above the minimum permissible pressure required to suppress cavitation. Hence, the minimum permissible pump inlet pressures of the engine are the key values in establishing the design of the pressurization system and propellant tanks. The pump inducers attached directly to the pump impellers (Figs. 2-19, 2-20, and 2-21) have been incorporated to reduce the required inlet pressure. In effect, the inducers act as a boost pump, adding head to the propellants to suppress cavitation in the main impeller. The engine pump minimum total inlet pressures are shown in Figs. 6-1 and 6-2. At nominal turbine speed and at a propellant temperatures of $60^{\circ}F$, the minimum permissible pump inlet pressures are 13.05 psia for fuel and 9.80 for the oxidizer.

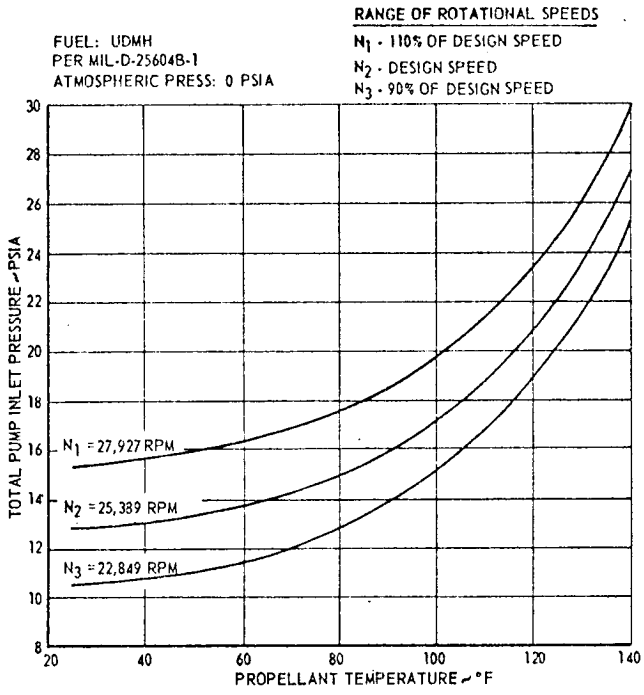


Fig. 6-1 Fuel Pump Minimum Inlet Pressure Versus Propellant Temperature

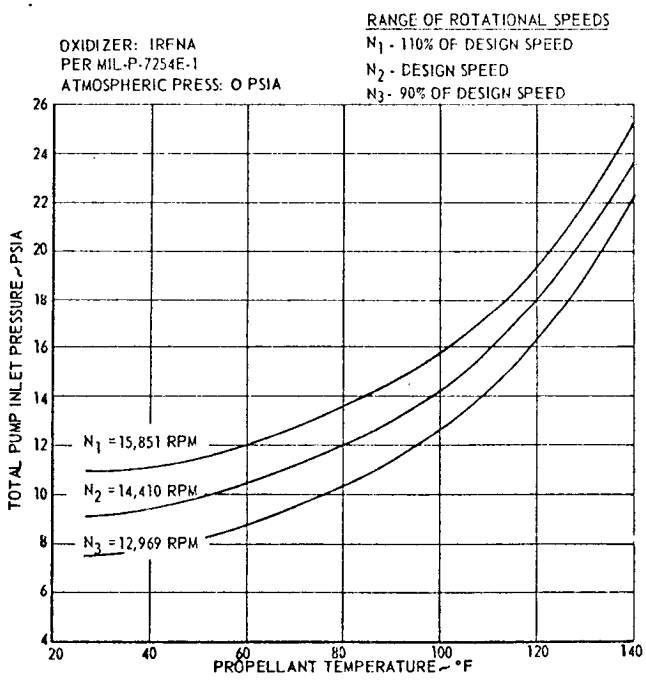


Fig. 6-2 Oxidizer Pump Minimum Inlet Pressure Versus Propellant Temperature

6.3 ORIFICE-FED PRESSURIZATION SYSTEM

The primary tool used for predicting orifice-fed pressurization system (OFPS) performance, and in part, the Gemini ATV primary propulsion system flight performance, is the computer program originally set up to determine feasibility. The capability of adequately predicting tank and sphere pressure-time histories was demonstrated at the Santa Cruz Test Base by PTV static firing tests.

The actual process involved in sphere blowdown is a comprise between adiabatic and isothermal. To compute these blowdown curves properly, a polytropic exponent, varying with time, was utilized.

A sufficient bias is established between the fuel and oxidizer tank (Fig. 6-3) to preclude common bulkhead reversal.

Examination of the starting transients (Fig. 6-3) indicates the following: The oxidizer and fuel tank pressures decrease initially because ullage volumes increase prior to actuation of the POHCV. At approximately 2 seconds after the control valve opens, the tanks repressurize to their maximum value and then decay gradually during first burn blowdown. After first burn shutdown, both tank pressures rise again. At 318 seconds after engine start signal, the oxidizer tank is isolated from the helium sphere. The fuel tank pressure continues to rise while oxidizer tank pressure remains constant; thus increasing fuel over oxidizer pressure differential, providing a greater safety margin against bulkhead reversal. During subsequent burns the tank pressures continue to blowdown until the mission requirements are completed. Tank 3-sigma variations indicate the bands in which the propellant tank pressures will remain during operation.

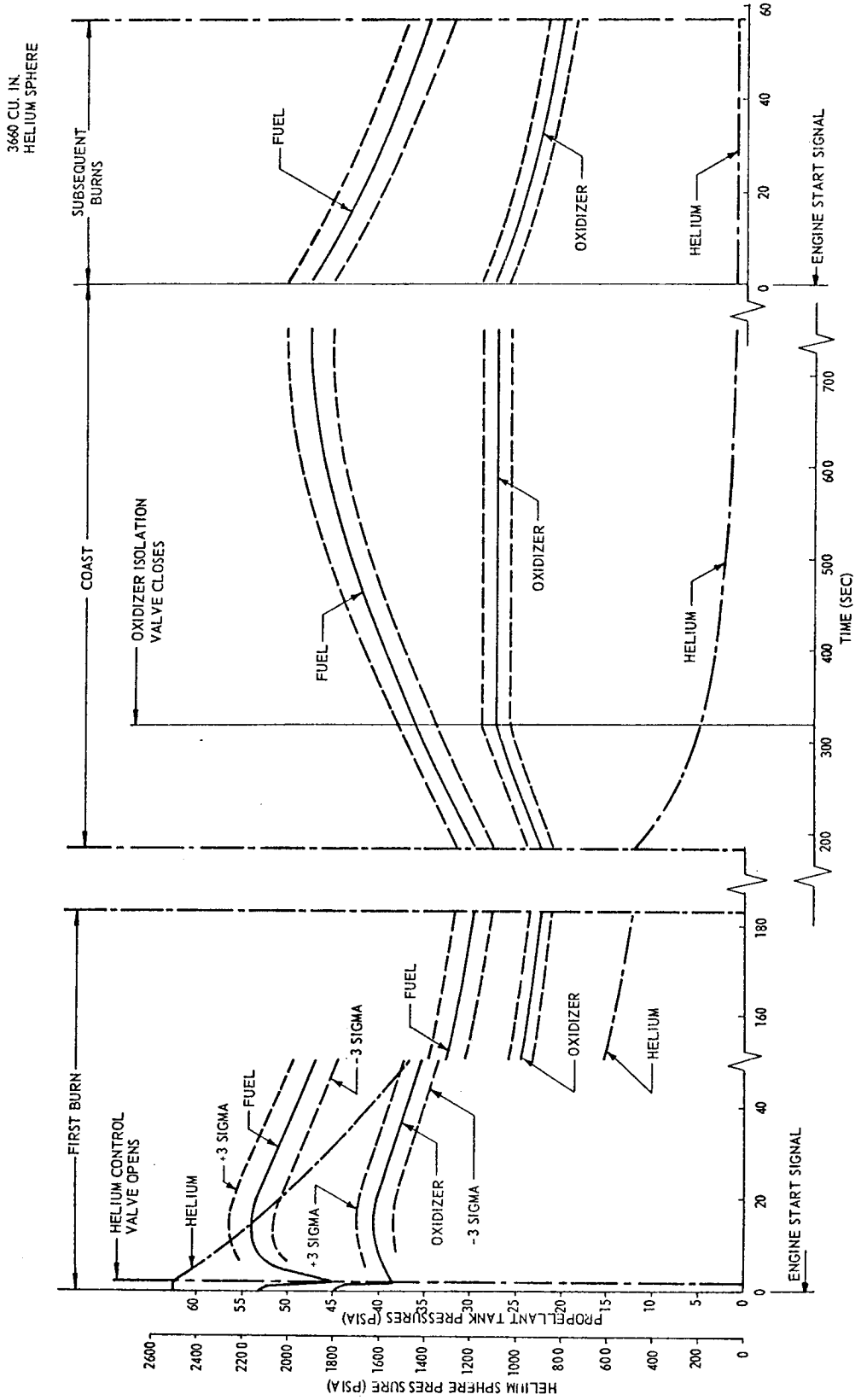


Fig. 6-3 Helium Sphere and Propellant Tank Pressure-Time Histories – Squib-Operated

6.4 SUBSYSTEM PERFORMANCE IN VACUUM FLIGHT

6.4.1 General

The propulsion subsystem performance in vacuum flight is defined as the performance of the engine integrated into the propulsion system and subjected to the varying pump inlet conditions which result from tank blowdown and from operation in the accelerating vehicle. Hence, vehicle weight is a parameter affecting engine performance and, therefore, the numerical performance values corresponding to subsystem performance will vary for individual flights. This difference results from the action of tank blowdown and acceleration on the pump inlet pressures, and consequently, on flow rates, mixture ratio, and thrust. The numerical values for the variation of these parameters are a function of vehicle weight and other operating conditions. For predicting the nominal flight performance of any specific vehicle, the engine performance, as obtained from the engine acceptance test data, is extrapolated to flight conditions using the engine-governing equations given in Paragraph 6.8. The parameters of thrust, propellant consumption, and mixture ratio are machine-computed as a function of time by an iterative process feeding back the effects of vehicle acceleration until propellant exhaustion occurs. To illustrate this procedure, the steps for calculating a sample point are outlined in the description which follows.

6.4.2 Determining Flight Performance Parameters

To determine the flight performance parameters of thrust, propellant consumption, and mixture ratio the operating conditions at the pump inlets and the turbine speed, as a function of these operating conditions, must be established.

The pump inlet operating conditions affecting the performance parameters of the engine are propellant density and pump inlet pressure. The effect of temperature, other than on density, has been empirically determined as negligible. For computing the nominal flight performance, propellants of nominal composition, and at a temperature of $60^{\circ}F$, are assumed.

The instantaneous pump inlet pressures for both pumps at any time are obtained from the following basic equation:

$$P_{inlet} = P_{tank} - \Delta P_{lines} + P_{acc}$$

where:

P_{tank} = the tank top pressure at the instant considered

ΔP_{lines} = the sum of the liquid and pneumatic line losses for fuel or oxidizer

P_{acc} = the acceleration pressure head of the propellants.

The line drops are design constants independent of the other variables. The acceleration pressure (P_{acc}) is a function of the liquid head (ft) above the pump inlet, the propellant density (lb/ft^3), and the vehicle acceleration (g's). It is related to these parameters by the following equation:

$$P_{acc} = \frac{\delta ha}{144g}$$

where:

δ = propellant density (lb/ft^3)

h = head (ft)

a = instantaneous acceleration

g = local gravitational constant (varies with altitude).

To determine the instantaneous value of h, the volume of propellants in the tank at the specified time and the relationship between the head and tank volume must be established. The volume of propellants in the tank is obtained by subtracting the volume of propellant consumed from the volume initially loaded in the tanks. The head-versus-volume relationship is a function of the geometry of the tank configuration. Typical curves for the integral tank are presented in Fig. 6-4.

The instantaneous acceleration is equal to the quotient of the thrust divided by the vehicle weight. As these parameters are interdependent, the value of acceleration must be determined by an iteration process. The procedure is to estimate a value for thrust and calculate the corresponding acceleration:

$$a = \frac{F}{W_v}$$

where:

W_v = vehicle weight

This value of acceleration is combined with the instantaneous head and propellant density to determine the acceleration pressure (P_{acc}). The acceleration pressure is substituted in the basic equation to determine the pump inlet pressure.

After the pump inlet pressures and propellant densities are determined, the turbine speed can be determined from the engine-governing equations given in Paragraph 6.8. The flow rates and thrust are then determined from the relationships also given in Paragraph 6.8.

This value of thrust is used to determine a new value for acceleration, and the calculation is iterated until the assumed thrust checks the derived thrust. Then, based on the computed flow rates and assumed time interval, succeeding points are calculated until propellant exhaustion or, in the case of a command shutdown, when the desired velocity increment is attained.

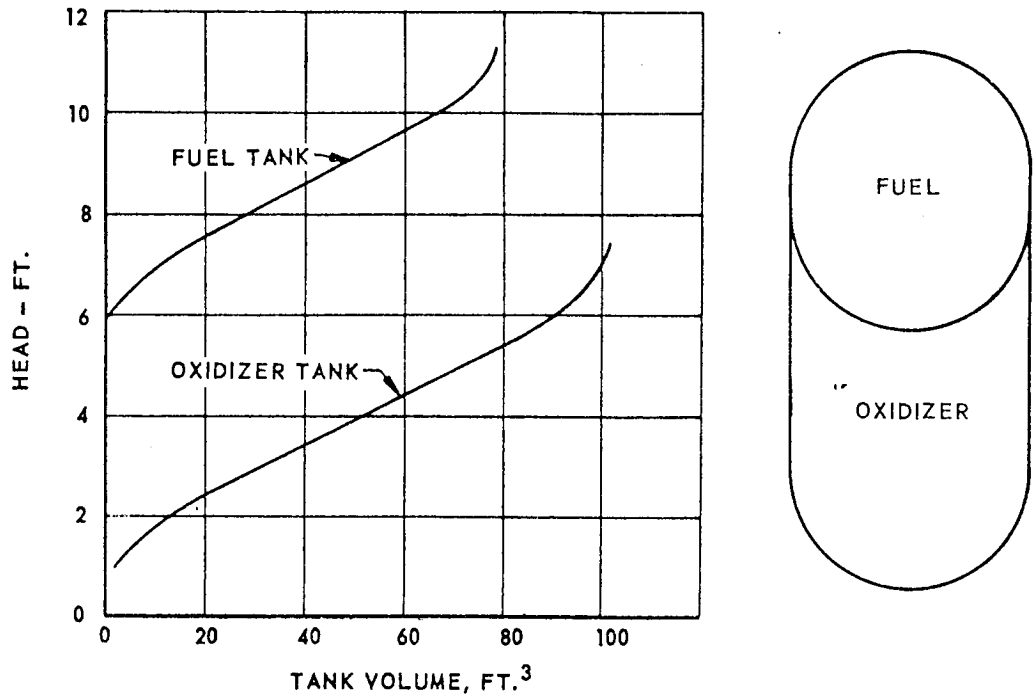


Fig. 6-4 Liquid Head Versus Volume for Propellant Tanks

6.4.3 Determination of Performance Capabilities

The criterion selected for determining performance capability is stage velocity increment as a function of payload. The velocity increment capability is based on minimum specific impulse. The payload capability is defined as the burnout weight less the weight of all of the propulsion system components and the propellant tank assembly. The entire calculation is based on the rocket equation for velocity increment assuming horizontal flight:

$$V = I_{sp}\, g \ln \frac{W_o}{W_o - W_p}$$

where:

W_o = burnout weight of vehicle

W_p = weight of propulsion system and propellant tanks

6.5 ENGINE VACUUM PERFORMANCE

The Subsystem B specification values for the performance parameter of thrust, burn time, specific impulse, and mixture ratio are presented in Table 6-1.

The predictability values shown in the table are the 3-sigma variations of vacuum performance for the primary engine parameters.

Table 6-1

ROCKET ENGINE PERFORMANCE AT ALTITUDE

Engine Performance Parameter	Rated	Limits of Set Performance		Predictability	Guaranteed Vacuum Performance Limits	
		Min.	Max.		Min.	Max.
Thrust, lb*	16,000	-2.5% 15,600	+2.5% 16,400	±3.08% ±493	-5.58% 15,107	+5.58% 16,893
Specific Impulse, sec**		291.5		±0.15% ±1.5	290.0	
Mixture Ratio, $\dot{W}_o/\dot{W}_f$	2.57	-1.5% 2.53	+1.5% 2.61	±0.6% ±0.015	-2.1% 2.515	+2.1% 2.625
Burn Time, sec	240.0					

*Figs. 6-5 and 6-6 present thrust variations with altitude.

**Fig. 6-7 presents I_{sp} change with altitude.

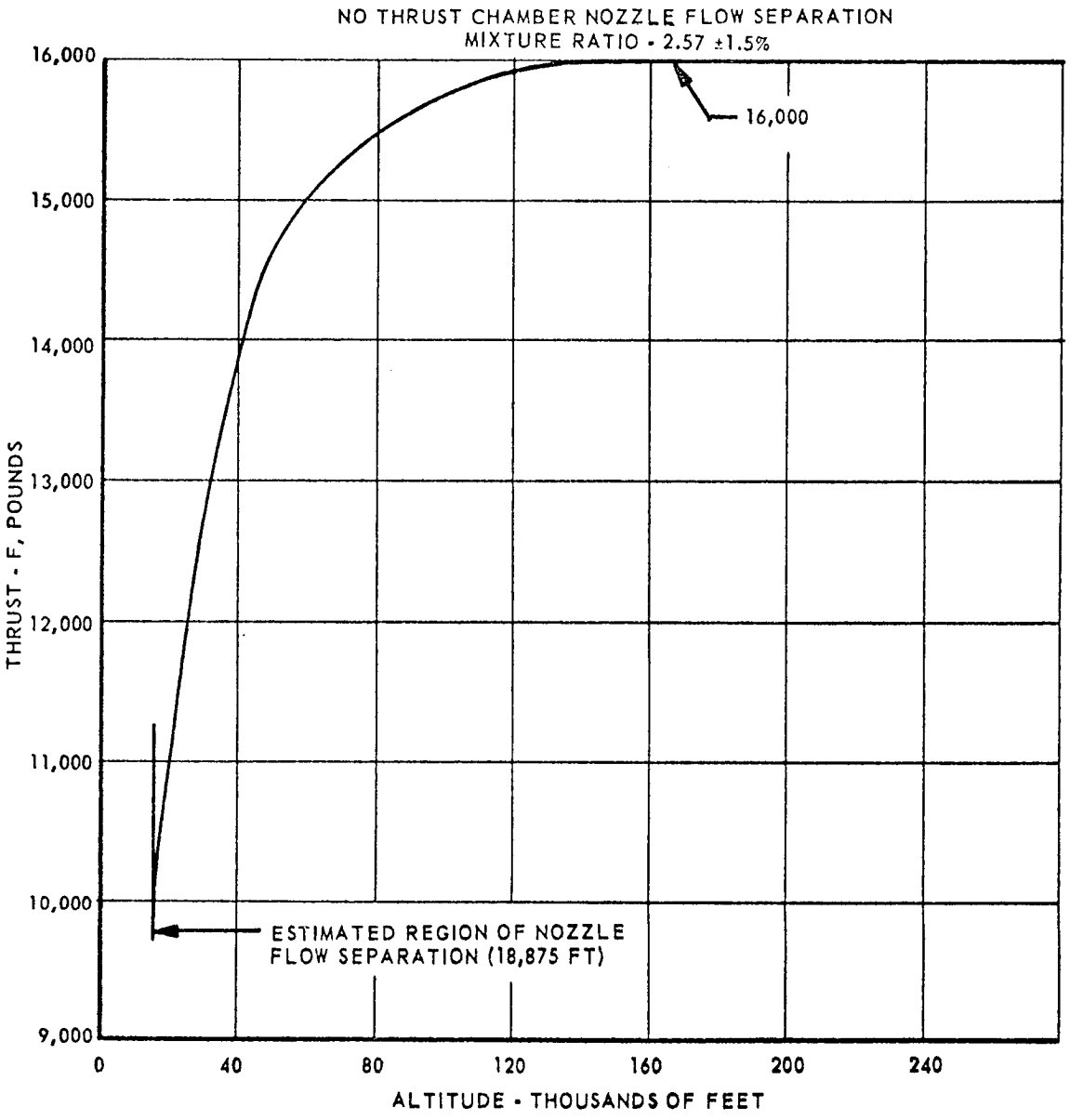
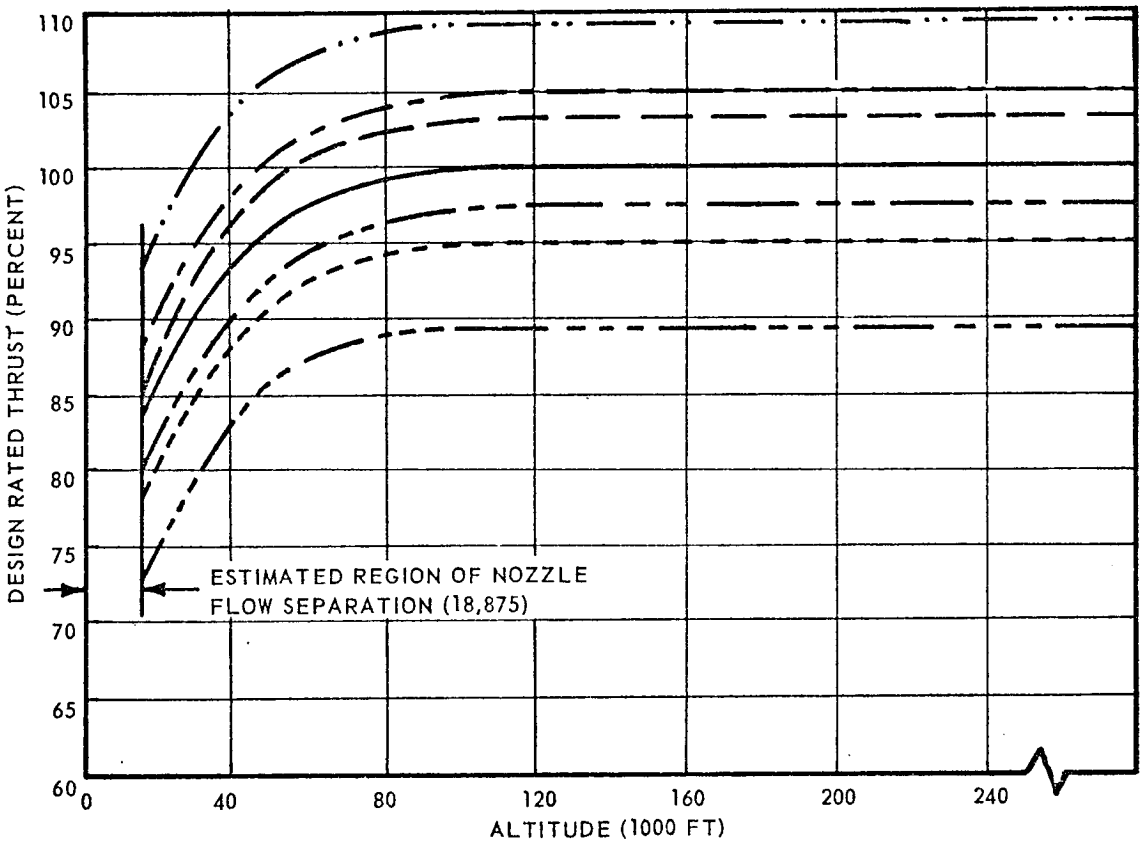


Fig. 6-5 Variation of Rated Thrust with Altitude



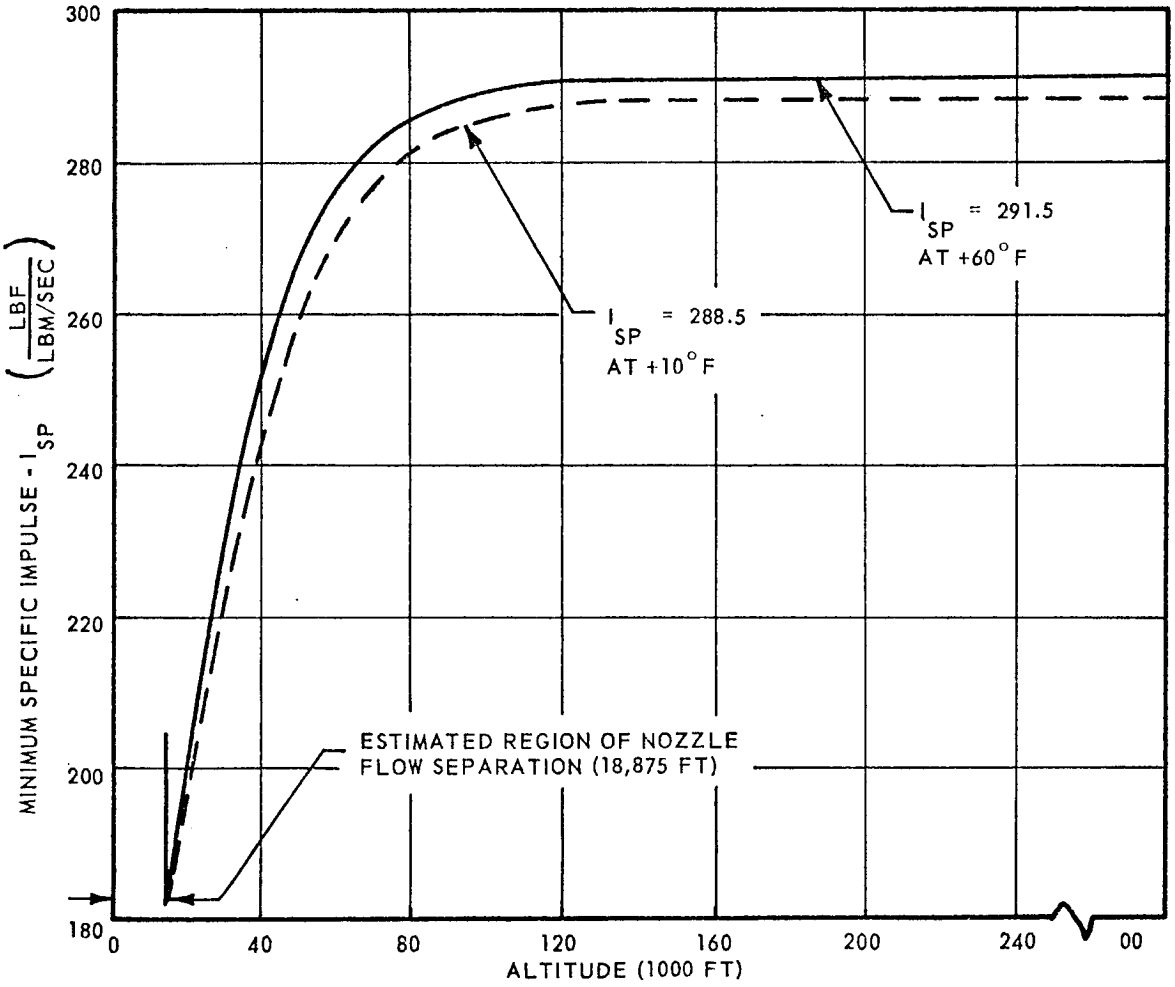
NOTE:

CURVES INCLUDE ONLY VARIATIONS DUE TO PROPELLANT DENSITY LIMITS, MANUFACTURING TOLERANCES, AND SPEED CONTROL AND NO THRUST-CHAMBER NOZZLE-FLOW SEPARATION.

LEGEND

- MAXIMUM AT +10°F
- MAXIMUM AT +32°F
- MAXIMUM AT +60°F
- RATED AT +60°F
- MINIMUM AT +60°F
- MINIMUM AT +100°F WITH PROPELLANTS AT +90°F
- MINIMUM AT +140°F

Fig. 6-6 Estimated Rocket Engine Thrust Limits Over Operating Requirements



NOTES:

NO THRUST-CHAMBER NOZZLE FLOW SEPARATION
MIXTURE RATION = 2.57 ±1.5 PERCENT

Fig. 6-7 Variation of Rocket Engine Minimum Overall Specific Impulse with Altitude

6.6 NOMINAL PERFORMANCE DATA

The engine model specification requires only the minimum acceptable value for specific impulse. Hence, the nominal specific impulse and the nominal flow rate for the engine are not specified values. The nominal value of specific impulse is an empirically derived parameter based on the average value obtained by a statistical evaluation of the data from a series of engine firings. As specific impulse is not a measurable parameter, it must be indirectly obtained by evaluating the parameters of thrust chamber characteristic velocity, c*, and nozzle thrust coefficient, C_F:

$$I_{sp} = \frac{c^* C_F}{g}$$

These parameters are obtained by measurements of thrust, flow rates, throat area, nozzle exit area, and chamber pressure as follows:

$$c^* = \frac{P_c A_t g}{\dot{W}_t}$$

$$C_F = \frac{F_t + P_a A_{ex}}{P_c A_t}$$

The average or nominal values for c* and C_F are 5,220 ft/sec and 1.829 respectively.

The minimum acceptable specific impulse for the XLR-81-BA 13 engine is 291.5 seconds. As engines currently being delivered are meeting this requirement, this figure may be considered nominal.

The nominal flow rate for the engine is defined as the rate which will result in nominal rated thrust at nominal specific impulse:

$$\dot{W}_T = \frac{F}{I_{sp}}$$

6.7 EXHAUST DUCT THRUST

The altitude value of the exhaust duct thrust can be computed as the sum of the gas momentum and the pressure force acting at the duct exit:

$$F_{ex} = (P_{ex} A_{ex}) + \left(\frac{\dot{W}_{gg}}{g} \right) \left(V_{ex} \right)$$

For sonic flow at the duct exit, the exhaust duct thrust will be 200 lb.

6.8 ROCKET ENGINE PERFORMANCE PREDICTION (PRE-FLIGHT)

In order to predict engine performance accurately, Gemini ATV variations from the standard Agena D must be taken into consideration. These variations are caused by:

- Nonstandard turbine exhaust duct length and shape
- Nonstandard hydraulic motor (gimbal) flow rates
- Nonstandard sphere size utilized in vehicle pressurization system
- Variation of propellant temperature.

A computer program combines these variations with previous flight experience and Bell Aerosystems Company (BAC) engine acceptance test data to predict flight performance.

Rocket engine performance at altitude can be calculated based on an average of the sea level corrected data presented in the BAC engine acceptance log book. Rocket engine performance at altitude is defined by the following averaged parameters:

- Mixture ratio at altitude $\overline{R}_{\infty}$
- Total mass flow rate at altitude $\overline{\dot{W}}_{\infty}$
- Thrust at altitude $\overline{F}_{\infty}$
- Specific impulse at altitude $\overline{I}_{sp\infty}$
- Turbine speed at altitude $\overline{N}_{\infty}$

Each of these parameters must be corrected from sea level to vacuum conditions with the exception of I_{sp}, which is assumed at a nominal 291.5. The thrust is derived analytically; i.e., using $\overline{\dot{W}}_{\infty}$, $\overline{I}_{sp\infty}$, and turbine exhaust thrust. Figures 6-8, 6-9, 6-10, and 6-11 show pump design characteristics.

6.8.1 Methods of Prediction

6.8.1.1 $\overline{N}_{\infty}$ ~ Turbine Speed at Altitude

$$\overline{N}_{\infty} = (K_n)\ (\overline{N}_{sl})$$

where:

$\overline{N}_{\infty}$ = turbine speed at altitude

K_n = computer constant

N_{sl} = turbine speed at sea level corrected. This value is available from BAC Engine Log Book.

6.8.1.2 $\overline{\dot{W}}_{\infty}$ ~ Mass Flow Rate at Altitude

$$\overline{\dot{W}}_{\infty} = K_{\dot{w}}\ \dot{W}_{sl}$$

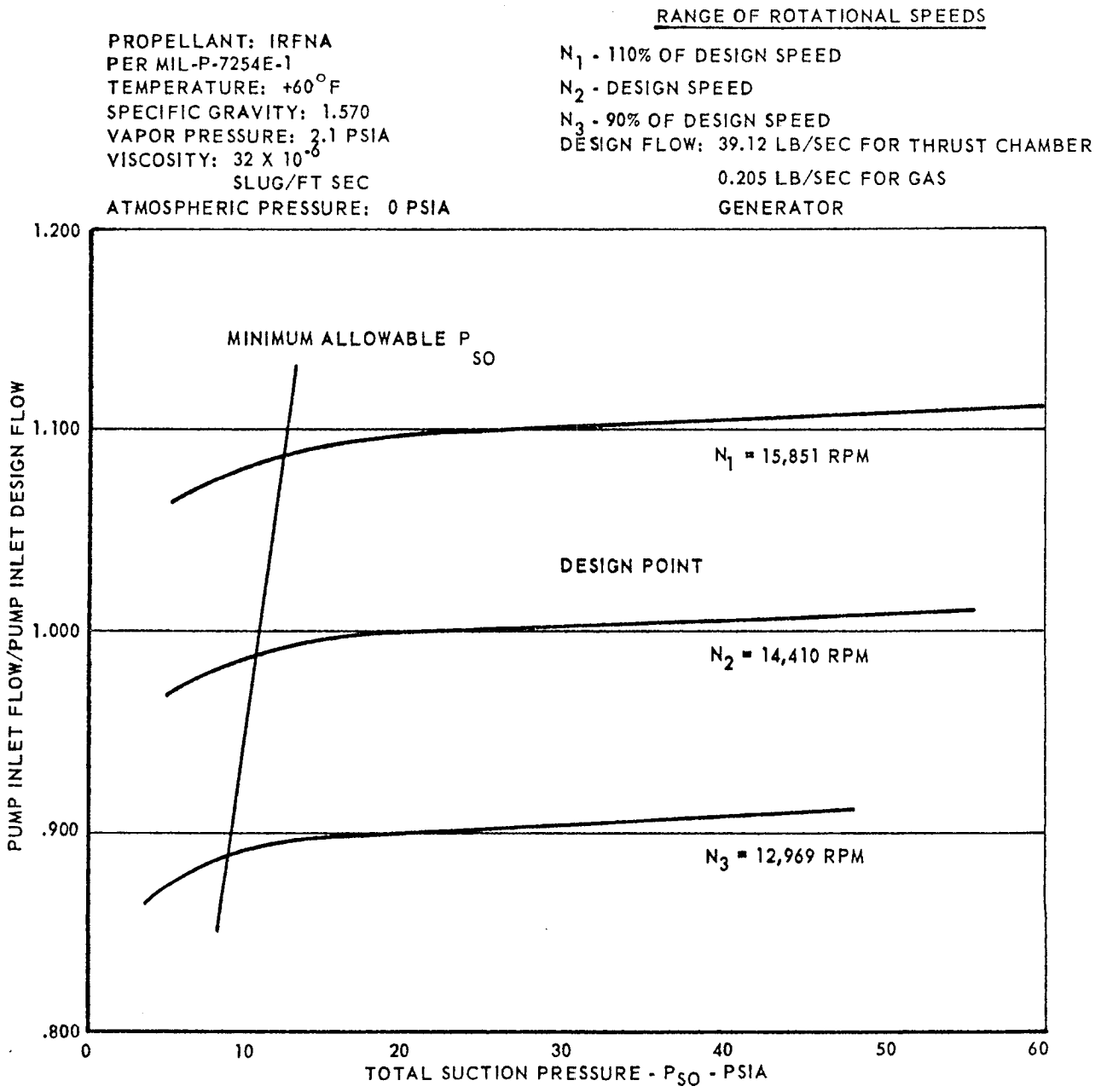


Fig. 6-8 Oxidizer Pump Flow Versus Total Suction Pressure

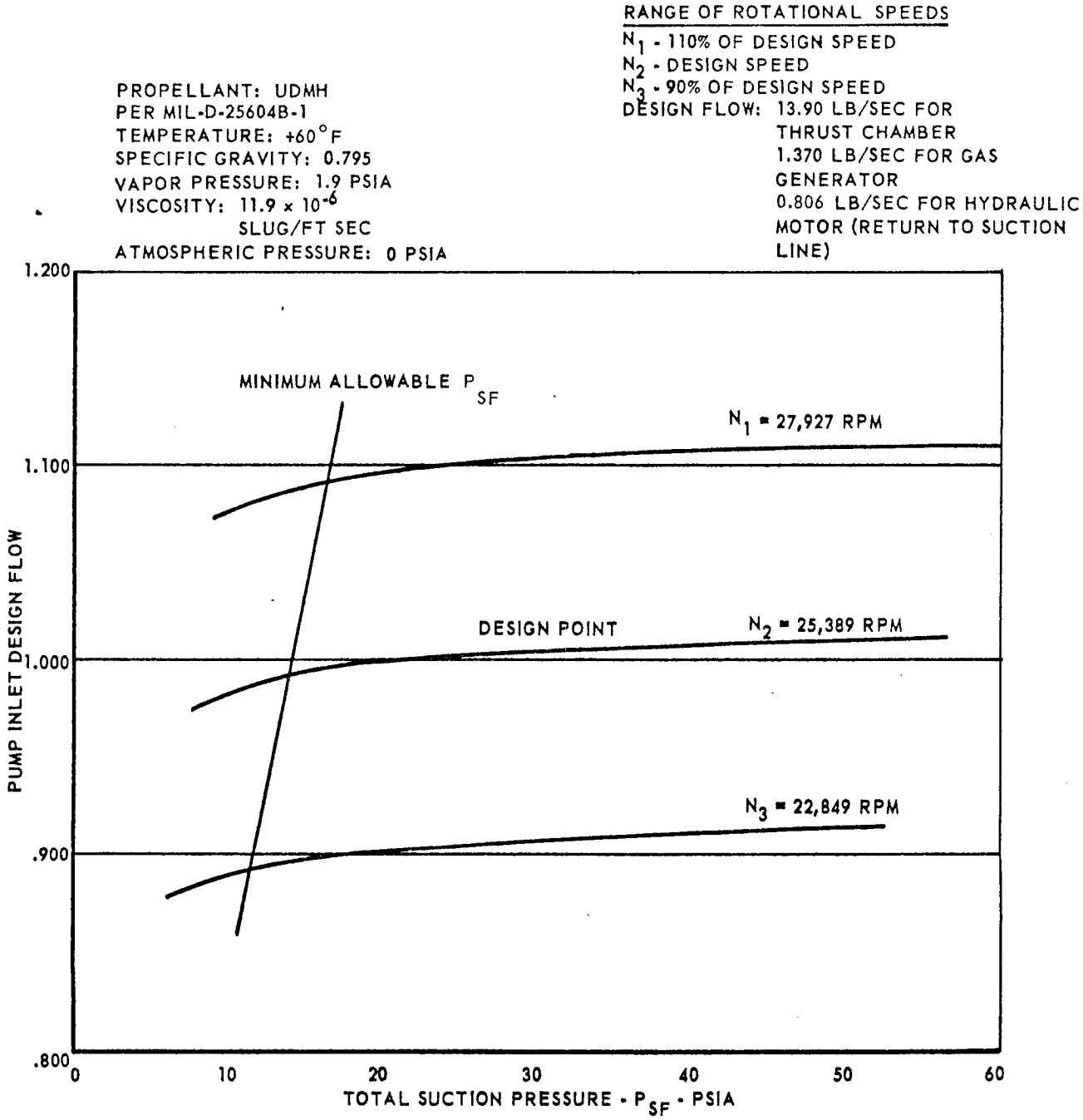


Fig. 6-9 Fuel Pump Flow Versus Total Suction Pressure

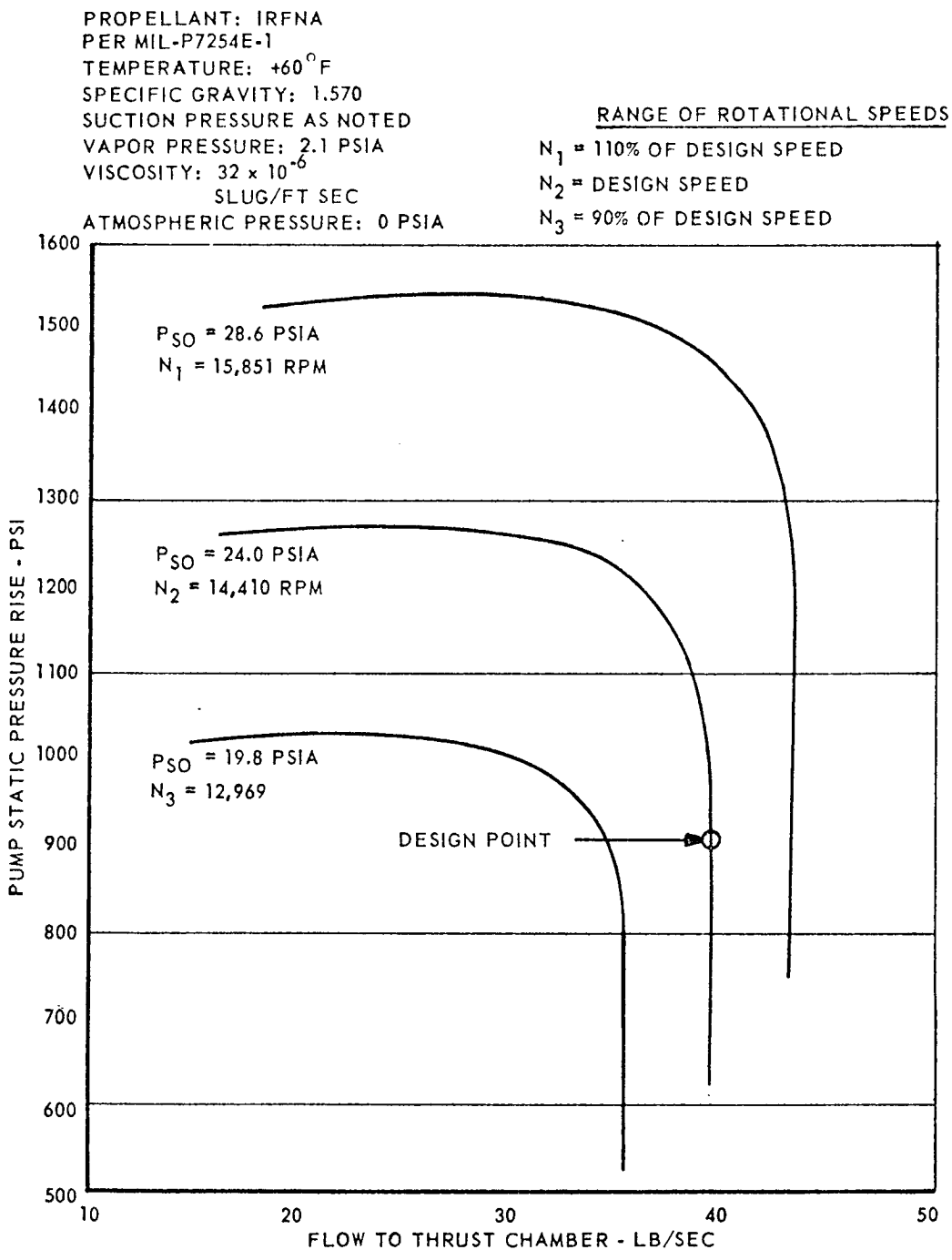


Fig. 6-10 Oxidizer Pump Static Pressure Rise Versus Flow Rate at Constant Speeds

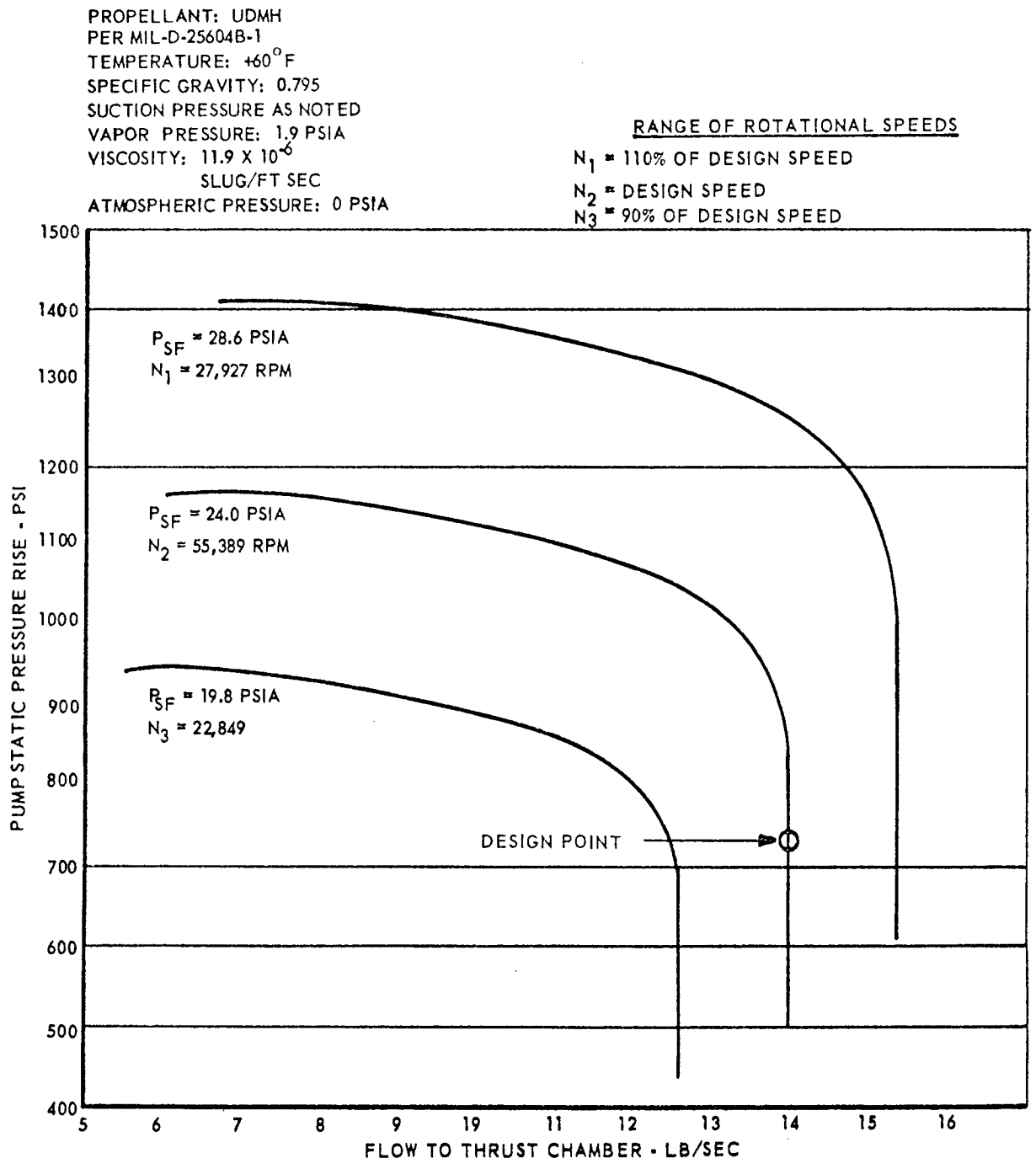


Fig. 6-11 Fuel Pump Static Pressure Rise Versus Flow Rate at Constant Speeds

where:

$\bar{\dot{W}}_{\infty}$ = mass flow rate at altitude

$\bar{K}_{\dot{w}}$ = computer constant

$\bar{\dot{W}}_{sl}$ = mass flow rate at sea level corrected. This value is available from BAC Engine Log Book.

6.8.1.3 $\bar{R}_{\infty}$ ~ Mixture Ratio at Altitude

$$\bar{R}_{\infty} = (R_{sl}) \ (K_r)$$

where:

$\bar{R}_{\infty}$ = mixture ratio at altitude

$\bar{R}_{sl}$ = average sea level corrected mixture ratio. This value is available in the BAC Engine Log Book.

$\bar{K}_r$ = computer constant

6.8.1.4 $\bar{F}_{\infty}$ ~ Thrust at Altitude

$$\bar{F}_{\infty} = (I_{sp\infty}) \ (\bar{\dot{W}}_{\infty}) + F_{exh}$$

where:

$\bar{F}_{\infty}$ = thrust at altitude

$\bar{I}_{sp\infty}$ = specific impulse at altitude. This is a fixed value of 291.5 $\frac{lbf}{lbm/sec}$

F_{exh} = exhaust duct thrust (constant 200 lb)

6.8.2 Example of Predicted Flight Values

Values taken from BAC Engine Log Book:

$$I_{sp} = 291.5 \frac{\text{lbf}}{\text{lbm/sec}}$$

$$N_{sl} = 24,497 \text{ rpm}$$

$$\dot{W}_{sl} = 53.34$$

$$R_{sl} = 2.55$$

Values taken from computer study:

$$K_n = 1.00208$$

$$K_{\dot{W}} = 1.06500$$

$$K_r = 1.02100$$

Solve for $\bar{N}_\infty$:

$$\bar{N}_\infty = (K_n)(N_{sl})$$

$$\bar{N}_\infty = (1.00208)(24,497)$$

$$\bar{N}_\infty = 24,548 \text{ rpm}$$

Solve for $\bar{\dot{W}}_\infty$:

$$\bar{\dot{W}}_\infty = (K_{\dot{W}})(\dot{W}_{sl})$$

$$\bar{\dot{W}}_\infty = (1.06500)(53.34)$$

$$\bar{\dot{W}}_\infty = 56.81$$

Solve for $\bar{R}_{\infty}$:

$$\bar{R}_{\infty} = (K_r)\ (\bar{R}_{sl})$$

$$\bar{R}_{\infty} = (1.02100)\ (2.55)$$

$$\bar{R}_{\infty} = 2.60 \text{ lb oxidizer/lb fuel}$$

Solve for $\bar{F}_{\infty}$:

$$\bar{F}_{\infty} = (I_{sp\infty})\ (W_{\infty}) + F_{exh}$$

$$\bar{F}_{\infty} = (291.5)\ (56.81) + 200$$

$$\bar{F}_{\infty} = 16,760 \text{ lb}$$

The predicted values for flight are determined to be:

$$\bar{N}_{\infty} = 24,548 \text{ rpm}$$

$$\bar{W}_{\infty} = 56.81$$

$$\bar{R}_{\infty} = 2.60 \text{ lb oxidizer/lb fuel}$$

$$\bar{F}_{\infty} = 16,760 \text{ lb}$$

$$\bar{I}_{sp\infty} = 291.5 \text{ (assumed nominal value)}$$

These values reflect typical and not actual predicted performance for any vehicle.

6.9 ROCKET ENGINE PERFORMANCE ANALYSIS (POST-FLIGHT)

In order to verify preflight data predictions, actual flight data are averaged over total steady-state burn time and analyzed to determine actual values for the following parameters:

- $\overline{\dot{W}}_{\infty}$
- $\overline{R}_{\infty}$
- $\overline{F}_{\infty}$
- $\overline{N}_{\infty}$

From these actual values, the previously assumed I_{sp} may be calculated.

6.9.1 Methods of Analysis

The following data are obtained from flight:

$\overline{P}_{so}$ = Oxidizer suction pressure, average value (psia).
(Measurement B-2 Oxidizer Pump Inlet Pressure)

$\overline{P}_{sf}$ = Fuel suction pressure, average value (psia).
(Measurement B-1 Fuel Pump Inlet Pressure)

$\overline{T}_{so}$ = Oxidizer suction temperature, average value (^{o}F).
(Measurement B-32 Oxidizer Pump Inlet Temperature)

$\overline{T}_{sf}$ = Fuel suction temperature, average value (^{o}F).
(Measurement B-31 Fuel Pump Inlet Temperature)

δ_o = Specific gravity oxidizer (Fig. 3-1).

δ_f = Specific gravity fuel (Fig. 3-1).

P_{vf} = Fuel vapor pressure (psi) (Fig. 2-22)

P_{vo} = Oxidizer vapor pressure (psi) (Fig. 2-22)

OVIP = Average oxidizer venturi inlet pressure (psia)
(Measurement B-11)

$\overline{FVIP}$ = Average fuel venturi inlet pressure (psia) (Measurement B-12)

$\overline{N}$ = Average turbine speed (rpm) (Measurement B-35)

$\overline{P}_c$ = Average thrust chamber pressure (psia) (Measurement B-91)

P_{velo} = Velocity head of oxidizer (constant 2.1 psia)

P_{velf} = Velocity head of fuel (constant 1.5 psia)

P_{to} = Total pressure of oxidizer (psia)

P_{tf} = Total pressure of fuel (psia).

The following equations are utilized in solving for the previously assumed parameters:

a. Equation 1:

$$P_{to} = P_{so} + P_{velo}$$

b. Equation 2:

$$P_{tf} = P_{sf} + P_{velf}$$

c. Equation 3:

$$H_{so} = (P_{to} - P_{vo}) \left(\frac{2.31}{\delta_o}\right) \left(\frac{24,000}{\overline{N}}\right)^2$$

where:

H_{so} = head at oxidizer pump inlet (feet)

d. Equation 4:

$$H_{sf} = (P_{tf} - P_{vf}) \left(\frac{2.31}{\delta_f}\right) \left(\frac{24,000}{\bar{N}}\right)^2$$

where:

H_{sf} = head at fuel pump inlet (feet).

The oxidizer and fuel pump inlet characteristics, C_o and C_f, are determined from Figs. 6-12 and 6-13.

e. Equation 5:

$$\bar{\dot{W}}_{tco} = (K_1) (C_o) (\delta_o) (\bar{N})$$

where:

$\bar{\dot{W}}_{tco}$ = oxidizer flow rate to thrust chamber

K_1 = constant (to be defined)

f. Equation 6:

$$\bar{\dot{W}}_{tcf} = (K_6) (C_f) (\delta_f) (\bar{N})$$

where:

$\bar{\dot{W}}_{tcf}$ = fuel flow rate to thrust chamber

K_6 = constant (to be defined)

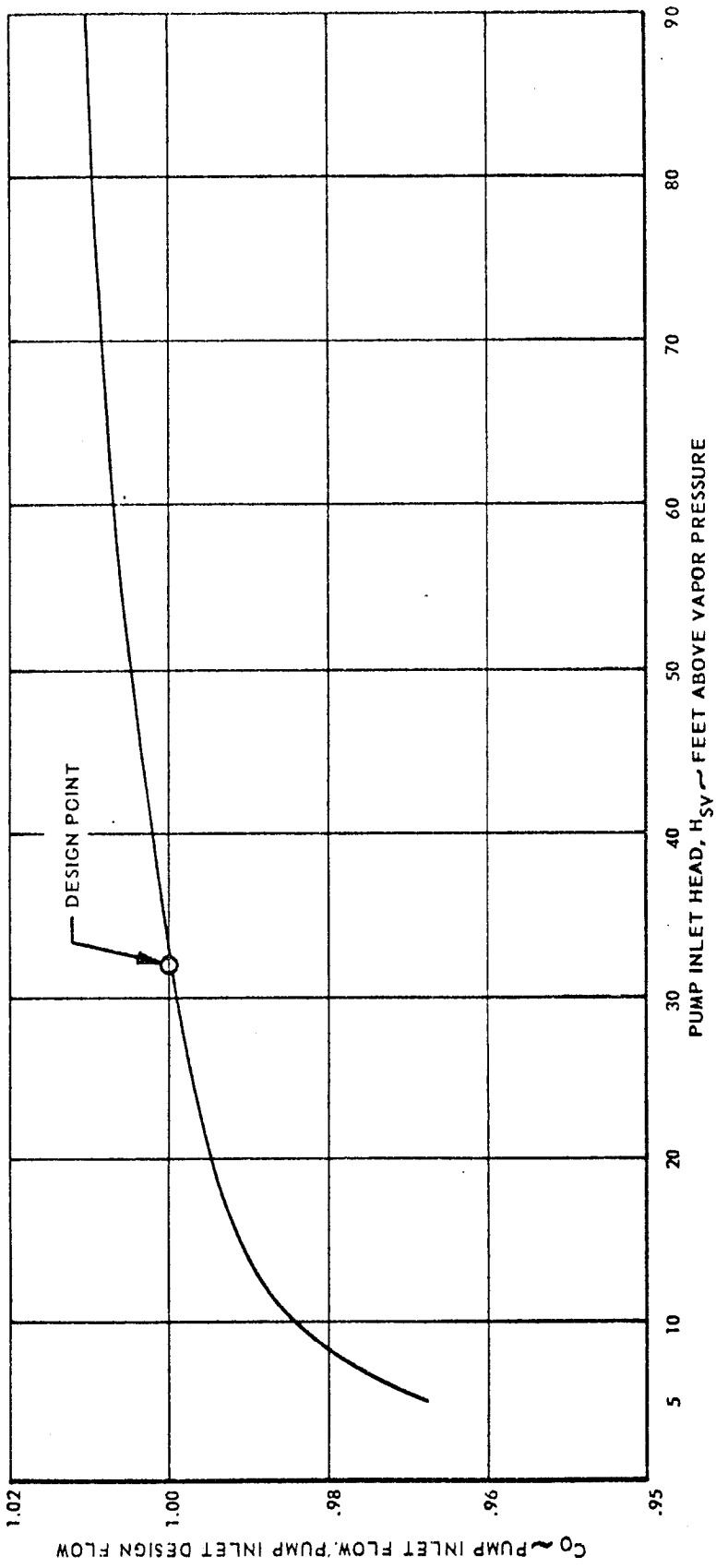


Fig. 6-12 Oxidizer Pump Flow Cavitation Characteristic

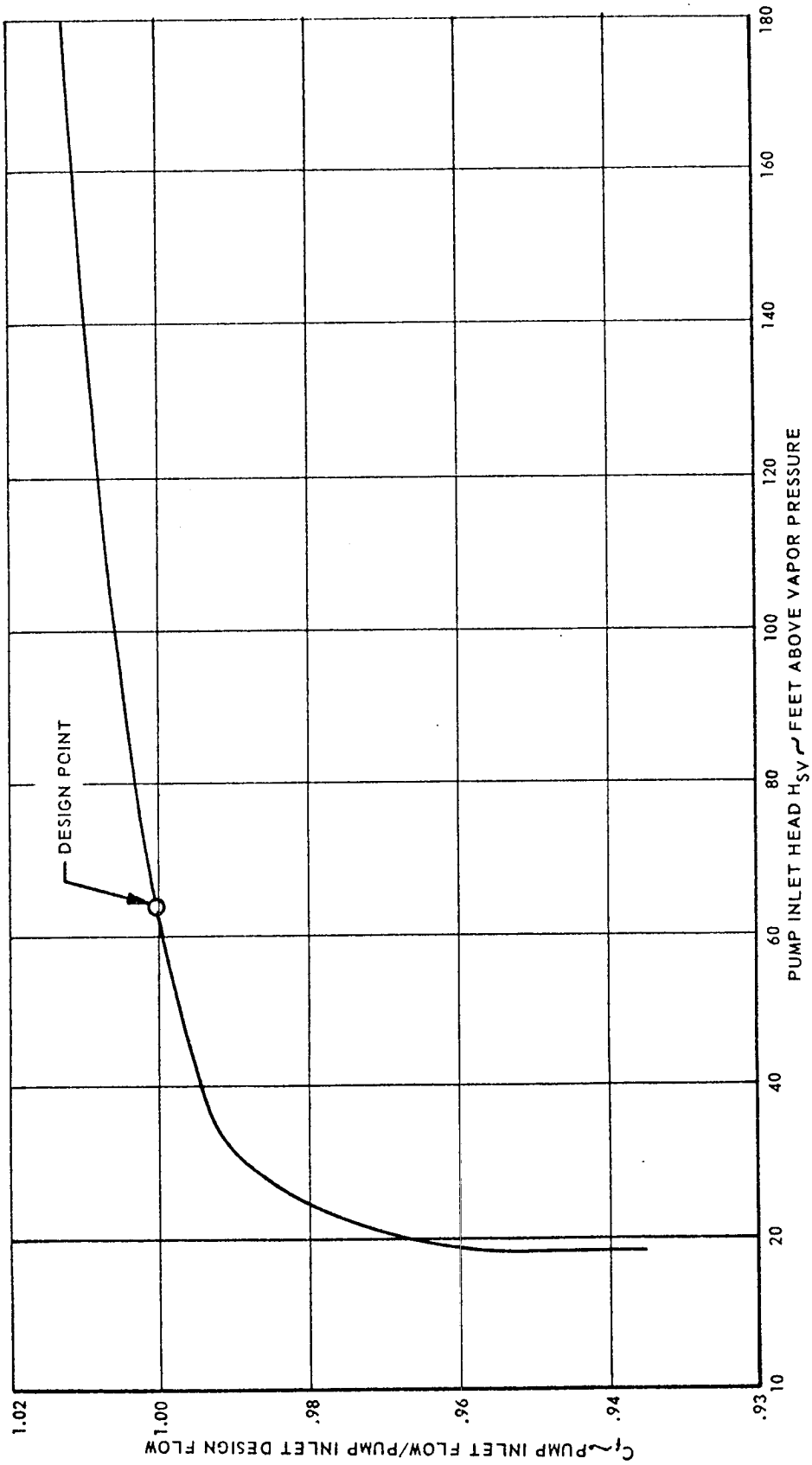


Fig. 6-13 Fuel Pump Flow Cavitation Characteristic

g. Equation 7:

$$\overline{\dot{W}}_{ggo} = (K_2) \sqrt{(\overline{OVIP}) \ (\delta_o - 0.003)}$$

where:

$\overline{\dot{W}}_{ggo}$ = flow rate of oxidizer to gas generator

K_2 = oxidizer venturi constant, obtained from BAC Engine Log Book

h. Equation 8:

$$\overline{\dot{W}_{ggf}} = (K_7) \sqrt{(\overline{FVIP}) \ (\delta_f - 0.004)}$$

where:

$\overline{\dot{W}_{ggf}}$ = flow rate of fuel to gas generator

K_7 = fuel venturi constant, obtained from BAC Engine Log Book

i. Equation 9:

$$\overline{G\dot{W}}_T = \overline{\dot{W}}_{ggf} + \overline{\dot{W}}_{ggo}$$

where:

$\overline{G\dot{W}}_T$ = total gas generator flow rate (average)

j. Equation 10:

$$\overline{TC\dot{W}_t} = \overline{\dot{W}_{tcf}} + \overline{\dot{W}_{tco}}$$

where:

$\overline{TC\dot{W}_t}$ = total thrust chamber flow rate (average)

k. Equation 11:

$$\overline{\dot{W}_\infty} = \overline{G\dot{W}_T} + \overline{TC\dot{W}_t}$$

l. Equation 12:

$$\overline{R_\infty} = \frac{\overline{\dot{W}_{ot}}}{\overline{\dot{W}_{ft}}}$$

where:

$$\overline{\dot{W}_{ot}} = \overline{\dot{W}_{tco}} + \overline{\dot{W}_{ggo}}$$

$$\overline{\dot{W}_{ft}} = \overline{\dot{W}_{tcf}} + \overline{\dot{W}_{ggp}}$$

m. Equation 13:

$$\overline{F_\infty} = (\overline{P_c})\ (C_f)\ (A_t) + F_{exh}$$

where:

C_f = thrust coefficient equals 1.829. This value is a constant and was obtained from a special test program.

A_t = throat area (in.^{2}). This value taken from BAC Engine Log Book

F_{exh} = turbine exhaust thrust. This value is constant (200 lb).

n. Equation 14:

$$\bar{I}_{sp} = \frac{\bar{F}_{\infty}}{\bar{\dot{W}}_{\infty}}$$

Definitions of the previously used constants (K_1 and K_6) are necessary:

K_1 – Defined:

$$K_1 = \frac{\bar{\dot{W}}_{ntco}}{(\delta_{on})\ (\bar{N}_n)\ (C_{on})}$$

where:

$\bar{\dot{W}}_{ntco}$ = mass flow rate of thrust chamber oxidizer at nominal rated conditions. This value is obtained from the BAC Engine Log Book.

δ_{on} = specific gravity of oxidizer at 60°F (1.570)

$\bar{N}_n$ = nominal turbine speed (24,000 rpm)

C_{on} = nominal pump coefficient, oxidizer (1.000)

K_6 – Defined:

$$K_6 = \frac{\bar{\dot{W}}_{ntcf}}{(\delta_{fn})\ (N_n)\ (C_{fn})}$$

where:

$\bar{\dot{W}}_{ntcf}$ = mass flow rate of thrust chamber fuel at nominal rated conditions. This value is obtained from the BAC Engine Log book.

δ_{fn} = specific gravity of fuel at 60°F (0.795)

C_{fn} = nominal pump coefficient, fuel (1.000)

6.9.2 Example of Flight Analysis

Flight Data (Average Value)		Measurement Giving Values
$\bar{P}_{so}$	= 14.30 psia	B-2
$\bar{P}_{sf}$	= 15.40 psia	B-1
$\bar{T}_{so}$	= 51.5°F	B-32
T_{sf}	= 53.0°F	B-31
δ_o	= 1.577	Fig. 3-1
δ_f	= 0.799	Fig. 3-1
P_{vo}	= 1.60	Fig. 2-22
P_{vf}	= 1.50	Fig. 2-22
$\overline{OVIP}$	= 1,040 psia	B-11
$\overline{FVIP}$	= 1,005 psia	B-12
$\bar{N}_\infty$	= 24,393 rpm	B-35
$\bar{P}_c$	= 504 psia	B-91
P_{velo}	= 2.1 psia	Constant
P_{velf}	= 1.5 psia	Constant

6.9.2.1 Solve for $\overline{N}_\infty$, $\overline{\dot{W}}_\infty$, $\overline{R}_\infty$, and $\overline{F}_\infty$

$\overline{N}_\infty$ - Solved:

$\overline{N}_\infty$ = average measured data obtained from flight records (Measurement B-35)

$\overline{N}_\infty = 24,393$ rpm

$\overline{\dot{W}}_\infty$ - Solved:

$$H_{so} = (P_{to} - P_{vo}) \left(\frac{2.31}{\delta_o}\right) \left(\frac{24,000}{\overline{N}_\infty}\right)^2$$

$$P_{to} = P_{so} + P_{velo}$$

$$P_{to} = 14.30 + 2.10$$

$$P_{to} = 16.40$$

$$H_{so} = (16.40 - 1.60) \left(\frac{2.31}{1.577}\right) \left(\frac{24,000}{24,393}\right)^2$$

$H_{so} = 20.99$ ft

From Fig. 6-12, $C_o = 0.995$

$$\overline{\dot{W}}_{tco} = (K_1)\ (C_o)\ (\delta_o)\ (\overline{N}_\infty)$$

$$K_1 = \frac{\overline{\dot{W}}_{tcon}}{(\delta_{on})\ (N_n)\ (C_{on})}$$

where:

$$\dot{\overline{W}}_{tcon} = 38.76 \text{ (from BAC Engine Log Book data)}$$

$$K_1 = \frac{38.76}{(1.570)\ (24,000)\ (1)}$$

$$K_1 = 0.001029$$

$$\dot{\overline{W}}_{tco} = (0.001029)\ (0.995)\ (1.577)\ (24,393)$$

$$\underline{\dot{\overline{W}}_{tco} = 39.34 \text{ lb/sec}}$$

$$H_{sf} = (P_{tf} - P_{vf}) \left(\frac{2.31}{\delta_f}\right) \left(\frac{24,000}{\overline{N}_\infty}\right)^2$$

$$P_{tf} = P_{sf} + P_{velf}$$

$$P_{tf} = 15.40 + 1.50$$

$$P_{tf} = 16.90$$

$$H_{sf} = (16.90 - 1.50) \left(\frac{2.31}{0.799}\right) \left(\frac{24,000}{24,393}\right)^2$$

$$\underline{H_{sf} = 43.09 \text{ ft}}$$

From Fig. 6-13, $C_f = 0.995$

$$\dot{\overline{W}}_{tcf} = (K_6)\ (C_f)\ (\delta_f)\ (N)$$

$$K_6 = \frac{\dot{\overline{W}}_{tcfn}}{(\delta_{fn})\ (N_n)\ (C_{fn})}$$

where:

$$\overline{\dot{W}}_{tcfn} = 13.75 \text{ (from BAC Engine Log Book data)}$$

$$K_6 = \frac{13.75}{(0.795)\ (24,000)\ (1)}$$

$$K_6 = 0.000721$$

$$\overline{\dot{W}}_{tcf} = (0.000721)\ (0.995)\ (0.799)\ (24,393)$$

$$\underline{\overline{\dot{W}}_{tcf} = 13.95 \text{ lb/sec}}$$

$$\overline{\dot{W}}_{ggo} = K_2 \sqrt{(\overline{OVIP})\ (\delta_o - 0.003)}$$

$K_2 = 0.004650$ from BAC Engine Log Book
(oxidizer venturi constant)

$$\overline{\dot{W}}_{ggo} = (0.004650) \sqrt{(1,040)\ (1.577 - 0.003)}$$

$$\underline{\overline{\dot{W}}_{ggo} = 0.188 \text{ lb/sec}}$$

$$\overline{\dot{W}}_{ggf} = K_7 \sqrt{(\overline{FVIP})\ (\delta_f - 0.004)}$$

$K_7 = 0.045060$ from BAC Log Book
(fuel venturi constant)

$$\overline{\dot{W}}_{ggf} = (0.045060) \sqrt{(1,005)\ (0.799 - 0.004)}$$

$$\underline{\overline{\dot{W}}_{ggf} = 1.273 \text{ lb/sec}}$$

Solving for total flow ($\dot{\overline{W}}_\infty$):

$$\dot{\overline{W}}_\infty = \dot{\overline{W}}_{tc} + \dot{\overline{W}}_{gg}$$

$$\dot{\overline{W}}_{tc} = 39.34 + 13.95$$

$$\dot{\overline{W}}_{tc} = 53.29$$

$$\dot{\overline{W}}_{gg} = 1.273 + 0.188$$

$$\dot{\overline{W}}_{gg} = 1.461$$

$$\dot{\overline{W}}_\infty = 53.29 + 1.461$$

$$\dot{\overline{W}}_\infty = 54.75 \text{ lbm/sec}$$

R_∞ (System) - Solved:

$$R_\infty = \frac{\dot{\overline{W}}_{ot}}{\dot{\overline{W}}_{ft}}$$

$$\dot{\overline{W}}_{ot} = \dot{\overline{W}}_{tco} + \dot{\overline{W}}_{ggo}$$

$$\dot{\overline{W}}_{ot} = 39.34 + 0.188$$

$$\dot{\overline{W}}_{ot} = 39.53$$

$$\dot{\overline{W}}_{ft} = \dot{\overline{W}}_{tcf} + \dot{\overline{W}}_{ggf}$$

$$\dot{\overline{W}}_{ft} = 13.95 + 1.273$$

$$\dot{\overline{W}}_{ft} = 15.22$$

$$R_\infty = \frac{39.53}{15.22}$$

$$R_\infty = 2.60 \text{ lb oxidizer/lb fuel}$$

F_∞ - Solved:

$$F_\infty = (P_c)\ (C_f)\ (A_t) + 200$$

where:

$A_t = 17.17 \text{ in.}^2$ from BAC Engine Log Book

$$F_\infty = (504)\ (1.829)\ (17.17) + 200$$

$$F_\infty = 15,828 + 200$$

$$F_\infty = 16,028 \text{ lbf}$$

6.9.2.2 Solve for new $\overline{I}_{sp\infty}$

$$\overline{I}_{sp\infty} = \frac{\overline{F}_\infty}{\dot{W}_\infty}$$

$$\overline{I}_{sp\infty} = \frac{16,028}{54.75}$$

$$\overline{I}_{sp\infty} = 292.75 \ \frac{\text{lbf}}{\text{lbm/sec}}$$

These values and the analysis are presented as an example and reflect typical rather than specific flight performance.

6.10 ESTIMATED ENGINE ALTITUDE PERFORMANCE

Table 6-2 is presented to show typical altitude performance and overall engine characteristics.

Table 6-2

ESTIMATED ENGINE ALTITUDE PERFORMANCE AND OVERALL CHARACTERISTICS

Parameter	Value
Characteristic Length	
Thrust Chamber	83 in.
Gas Generator	98 in.
Characteristic Velocity	
Thrust Chamber	5,220 ft/sec
Gas Generator	3,375 ft/sec
Flow Rate	
Oxidizer	
Thrust Chamber	39.12 lb/sec
Gas Generator	0.205 lb/sec
Fuel	
Thrust Chamber	13.90 lb/sec
Gas Generator	1.370 lb/sec
Hydraulic Motor Pump	6.75 gpm
Mixture Ratio	
Engine	2.57
Thrust Chamber	2.81
Gas Generator	0.150

Table 6-2 (Cont)

Parameter	Value
Pressure (Static)	
Thrust Chamber	506 psia
Thrust Chamber Nozzle Extension Exit	1.87 psia
Gas Generator Chamber	485 psia
Turbine Manifold	475 psia
Turbine Manifold Nozzle Exit	12.6 psia
Turbine Exhaust Exit	7.8 psia
Specific Impulse	
Engine (minimum)	291.5 $\frac{\text{lbf}}{\text{lbm/sec}}$
Thrust Chamber (minimum)	296.5 $\frac{\text{lbf}}{\text{lbm/sec}}$
Thrust	
Engine	
Normal Steady State	16,000
Thrust Chamber	15,800
Thrust Coefficient	
Engine (Thrust Chamber 45:1)	1.829
Turbine Pump Speeds	
Turbine	
Normal Steady State	24,800 rpm
Oxidizer Impeller	14,410 rpm
Fuel Impeller	25,389 rpm
Velocity	
Thrust Chamber Nozzle Extension Exit	8,727 ft/sec
Gas Generator Throst	575 ft/sec

Table 6-2 (Cont)

Parameter	Value
Area	
Thrust Chamber Throat	17.12 $in.^2$
Thrust Chamber Nozzle Exit	227.4 $in.^2$
Thrust Chamber Nozzle Extension Exit	770.7 $in.^2$
Turbine Exhaust Duct Exit	12.6 $in.^2$
Turbine Manifold Nozzle Throat	0.058 $in.^2$
Turbine Manifold Nozzle Exit	0.283 $in.^2$
Leakage	
Pump Primary Seal	
Fuel (maximum)	500 cc
Oxidizer (maximum)	500 cc
Pilot Operated Solenoid Valve Overboard (maximum)	85 cc
Pressures	
Fuel	
Pump Suction (total pressure)	24 psig
Pump Discharge	
Gas Generator	1,045 psia
Hydraulic Motor Pump Without Orifice	1,045 psia
Thrust Chamber	750 psia
Gas Generator Injector Manifold	615 psia
Drop Across Gas Generator Injector	130 psi
Thrust Chamber Injector Manifold	600 psia
Drop Across Thrust Chamber Injector	94 psia

Table 6-2 (Cont)

Parameter	Value
Oxidizer	
Pump Suction (total pressure)	24 psia
Pump Discharge	
Gas Generator	1, 105 psia
Thrust Chamber	950 psia
Gas Generator Injector Manifold	570 psia
Drop Across Gas Generator Injector	85 psia
Thrust Chamber Injector Manifold	600 psia
Drop Across Thrust Chamber Injector	95 psia
Speed (rpm)	
Turbine Critical Speed	30, 750 rpm
Temperature	
Hot Gas	
Gas Generator	1, 400°F
Thrust Chamber	4, 450°F
Skin	
Gas Generator (Chamber)	200° to 600°F
Thrust Chamber (Convergent Nozzle) (maximum)	590°F
Thrust Chamber Nozzle Extension (maximum)	2, 300°F
Turbine Manifold	1, 000° to 1, 200°F
Fuel	
Pump Inlet	60°F
Gas Generator Injector	70°F
Thrust Chamber Injector	70°F
Oxidizer	
Pump Inlet	60°F
Gas Generator Injector	65°F
Thrust Chamber Injector	65°F

Section 7
PYROTECHNICS

7.1 INTRODUCTION

Ordnance devices used in the Gemini-Agena Target Vehicle (ATV) perform some of the most critical flight functions. Ballistic units have been chosen because of the history of high reliability and rapid actuation time inherent in developed ordnance items.

7.2 DEVICES USED

The pyrotechnic devices used in the Gemini ATV are listed in Table 7-1.

7.2.1 Pressure Cartridges

Two types of pressure cartridges, LMSC-1062363-3 (M-69) and 1463174-1 (M-11), are employed in the Gemini Program to actuate pyrotechnic devices. Functionally, the two units differ only in pressure pulse output. The method of construction is slightly different in each case.

Cartridge LMSC-1062363-3 (M-69) — The M-69 is capable of producing 6000 psi minimum in a 5-cc container and is employed in the first operation of the helium control valve.

The cartridge is constructed with a modified AND 10056-5 threaded output end, 0.559-in. long. It is approximately 1.09-in. long, 0.5-in. in diameter, and has a 0.625 in.-diameter hexagon in the center for wrench flats. The connector end is manufactured to mate with the Bendix Pygmy PT06E-8-4S bayonet type connector. Internally, the M-69 has four connecting pins

Table 7-1
GEMINI PYROTECHNICS

LMSC Part No.	Nomenclature	Quantity	Where Used
	Standard Agena D		
1062363-3	Type M-69 Squib	2	POHCV*
1463174-1	Type M-11 Squib	12	POHCV*, Horizon Sensor Panels Pin Pullers and Pin Pushers
1395698-1	Separation Charge	1	Booster Adapter
1396224-501	Separation Detonator	1	Booster Adapter
1347107-503	Self Destruct Charge	1	Booster Adapter
1312289-503	Self Destruct Initiator	1	Booster Adapter
1062410-5	Booster Retrorocket	2	Booster Adapter
	Program Peculiar		
1354376-501	Separation Bolt Assembly	4	———
1461809	Detonating Cartridge	8	For Separation Bolt Assembly

*POHCV – Pyrotechnically operated helium-control valve

(to match the Bendix connector) contained within four individual glass seals fused to individual holes through the cartridge body. A 0.003-in.-diameter nichrome bridgewire is soldered across each set of two pins, A to D and B to C. A bridgewire priming mix, composed of lead styphnate and a desensitizer additive, encapsulates the wire ends and is subsequently

surrounded by a circular phenolic sleeve. Approximately 227 mg (3-1/2 grains) of Hercules Bullseye powder are loaded on top of the priming mix to form the main charge. The output end of the cartridge is sealed with a 0.006-in.-thick lead closure disc which is cemented in place with NP 428 adhesive. The cartridge end is crimped over to retain the seal assembly and then coated with Poly-Ep-Platon epoxy compound to complete the closure process. Table 7-2 presents the electrical characteristics of the cartridge.

Table 7-2
CARTRIDGE ELECTRICAL CHARACTERISTICS

Maximum Test Current	10 ma per bridge
Resistances:	
Bridge Circuit	0.45 to 0.85 ohm at 70.0° ±20°F
Any Pin-to-Case	1 meg minimum
Bridge-to-Bridge	1 meg minimum
Maximum Safe Current	0.5 amp per bridge for 5 min @ 26.0 ±2.0 vdc
All-Fire Current	5.0 amp per bridge for 15 ms @ 26.0 ±2.0 vdc

Cartridge LMSC-1463174 (M-11) — The M-11 cartridge is capable of producing 3000 psi minimum in a 2-cc closed volume of 70° ±20°F. This M-11 cartridge is used in the second operation of the pyrotechnically operated helium-control valve (POHCV). The M-11 cartridge is constructed with a modified AND 10056-5 threaded output end, 0.300-in. long. The tapered portion of the standard AND has been removed to allow the output end of the cartridge to be efficiently sealed. This cartridge is approximately 0.840-in. long and 0.5-in. in diameter, and has a 0.625-in.-diameter hexagon in the center for wrench flats. The connector end is manufactured to mate with the Bendix Pygmy PC06E-8-4S screw-on type connector. Bridgewire connecting pins are sealed with separate glass seals fused to individual holes in the cartridge body.

A 0.003-in.-diameter nichrome bridgewire is soldered across each set of two pins, A to D and B to C. A bridgewire primary mix, composed of zirconium and barium chromate, surrounds each of the two wires, and 65 mg of Hercules Bullseye powder are loaded around and on top of the primary mix to form the main charge. The output end of the cartridge is covered by a 0.006-in.-thick lead foil closure sealed in place with a low-melting-point solder. Table 7-2 gives the electrical characteristics of the cartridge.

7.2.2 Booster Adapter Separation

Separation Charge – The separation charge provides the energy to fracture the booster-Agena joint. This separation charge is a mild detonating fuse (MDF), which is a 10 grain per foot RDX explosive core, contained by a flexible lead-antimony sheath and covered by a thin polyethylene extruded coating. The MDF is used in the separation system because of the following highly desirable properties:

1. High detonation rate, approximately 20,000 fps which guarantees almost instantaneous separation
2. Controlled predictable explosive power which prevents inadvertent damage to surrounding components or structure
3. Relative insensitivity to shock friction, heat, and stray electrical energy which reduces the hazards in handling, storing, and installation.

The RDX core is identified by a maximum of 1-percent pink dye molded into the material, which provides for easy detection and disposition in the event of any inadvertent spillage. Sheathing alloy is composed of 7-percent antimony in a lead base to increase the strength, and a polyethylene cover to provide waterproofing in case of sheathing porosity.

Fracture Joint – A frangible ring section is constructed with three closely adjacent parallel grooves milled around the inside circumference of the booster adapter. In addition, three longitudinal grooves which cut across the circular grooves are placed 120-deg apart around the inside of the ring section. Detonation of the fuse fractures the joint, thus producing a gap between the booster adapter and Agena. Six small strips of the adapter material are blown out and away from the vehicle.

Detonating Cord Ring – An extruded-aluminum detonating-fuse ring is fastened over the three parallel grooves. This ring contains the fuse, provides backup for the explosive reaction, and focuses the detonation pressure on the grooved section of the frangible joint.

Separation Detonator – The separation detonator assembly consists of two LMSC-1062876 electric detonators positioned in a detonator holder and restrained by Locktite sealant compound. These detonators receive a firing stimulus which heats up a lead-styphnate-coated bridgewire. The lead styphnate reacts to the heat produced by the bridgewire and detonates; this reaction is transferred to the adjacent lead azide which also detonates and initiates the main charge of RDX. Firing of the two detonators initiates the firing of the ends of the detonating fuse protruding through the collets. Table 7-3 presents the electrical characteristics of the detonators.

Table 7-3
SEPARATION DETONATOR ELECTRICAL CHARACTERISTICS

Maximum Test Current	10 ma at 50 vdc
Resistances:	
Pins A to D	0.90 to 1.10 ohms
Pins C to B	0.90 to 1.10 ohms
Pins A to C	1 meg minimum
Pin A to Case	1 meg minimum
Maximum No-Fire Current	0.50 amp for 5 min
Minimum Sure-Fire Current	1.5 amp for 6.5 min

7.2.3 Vehicle Destruct Systems

The vehicle incorporates two separate destruct systems: self-destruct and command destruct. The self-destruct system provides a means of destroying the vehicle during the boost phase of flight, in the event that an uncontrollable condition occurs wherein continued flight would constitude a hazard to property and personnel. The command destruct system is used only for control purposes. In this application, the pyrotechnic devices are removed from the system. The command receiver and associated equipment a re used to transfer a shutdown signal to the engine.

7.2.3.1 Self-Destruct System. This system consists principally of a shaped explosive destruct charge, a destruct charge initiator, four auxiliary destruct batteries, two premature separation switches (also known as destruct lockout switches), and related destruct system circuitry. All system components are installed in the satellite adapter section. The destruct charge, initiator, and batteries are adjacent to each other on the outside of the adapter, just aft of the vehicle separation plane. The two premature separation (destruct lockout) switches are inside the adapter.

The self-destruct system is capable of being actuated by a command signal from the Range Safety Officer or automatically upon premature separation of the Agena-D vehicle and booster. This system is electrically connected with the first-stage booster destruct system to provide the capability for simultaneous destruction of both vehicles by the ground command signal transmitted to the booster destruct system. Destruction may be effected any time between liftoff and booster thrust cutoff. Safety features are incorporated into the system to prevent accidental detonation of the destruct charge (1) during prelaunch checkout on the launch pad and (2) during flight until the normal separation of the Gemini-ATV and booster.

Shaped Charge — the flight termination system includes an explosive shaped charge (Figs. 7-1 and 7-2) and destruct initiator (Fig. 7-3). This assembly (Fig. 7-4) is mounted in the booster adapter and so aligned that the center-line of the charge and initiator assembly pass through the standpipe in the propellant tank and the common diaphragm separating the fuel from the oxidizer.

Destruction of the Agena vehicle results when the tank stand pipe is severed and the diaphragm is ruptured, thus allowing the fuel and the oxidizer to mix. This is accomplished by firing the shaped or coned charge. Since the propellants are hypergolic, vehicle destruction is instantenous upon mixture. By using this method, in which a metal cone is backed by a high explosive and is concentrated or focused into a metallic jet on firing, the necessary penetrating power required to pierce the tank, penetrate the oxidizer, and rupture the diaphragm is achieved.

The shaped charge assembly is 3.600-in. in diameter and 5.33-in. in length and weighs 8.5 lb. Of this total weight, approximately 1.3 lb is composition A-3 high explosive. The dual booster pellets are cased RDX, 0.590-in. diameter. RDX is also the explosive material used in the relay pellet, which is 1.480-in. in diameter and 0.75-in. thick. All the explosive components — charge, booster, and relay — are pressed items which are bonded into the case with an epoxy compound. The copper cone, which has a 60-deg angle with a 0.580-in. diameter flat at the apex, is also bonded to the charge. The edge of the case is crimped over to hold the whole assembly securely in place.

Initiator — The explosive initiator assembly (Fig. 7-3) contains three fuse-train elements. The first element is separated from the third by a steel rotor; this rotor contains one hole which either aligns with (arms) or disaligns with (makes safe) the explosives. The rotor hole contains the double-end second element. When this element is rotated to the arm position, the explosive train from the first to the third element is completed.

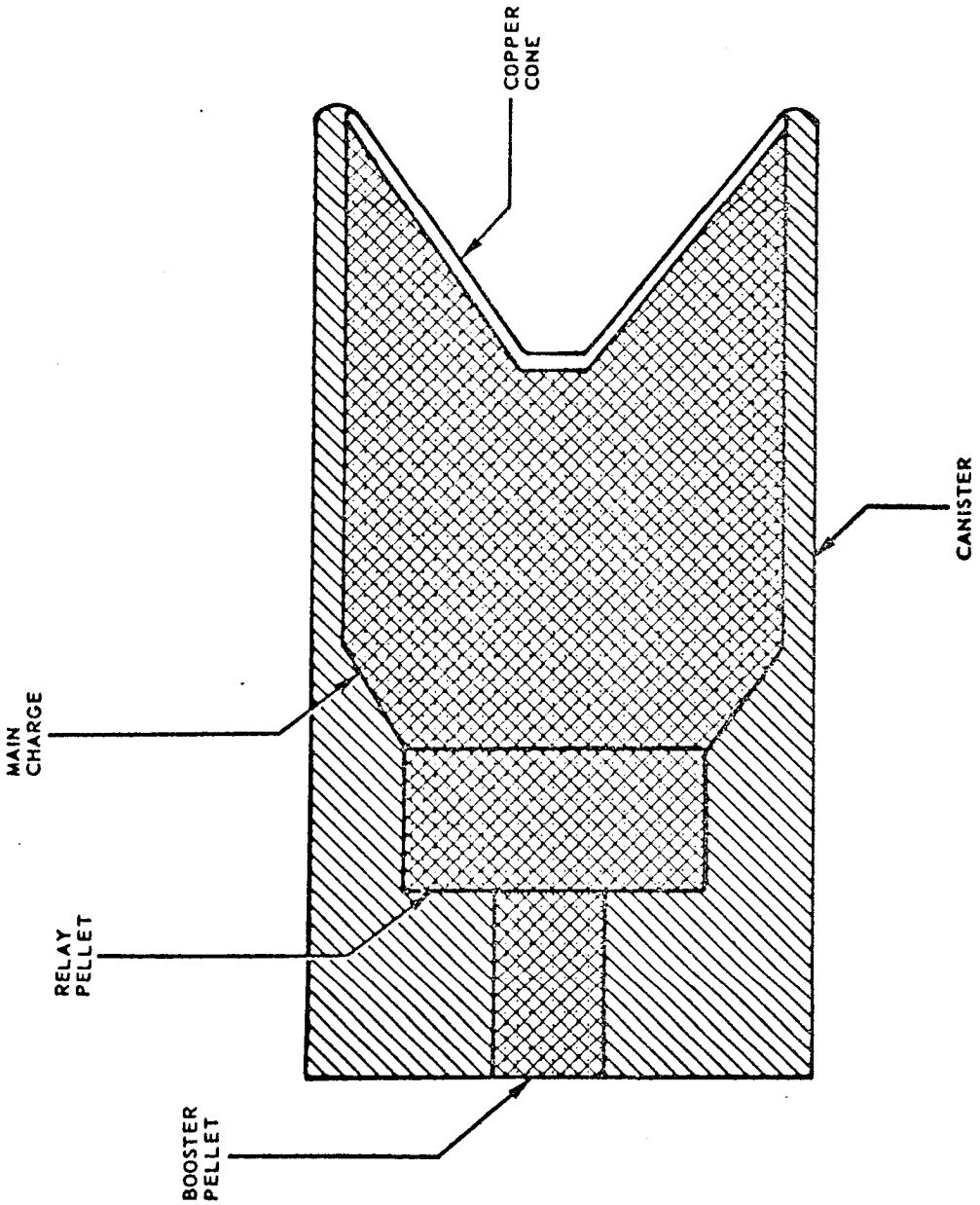


Fig. 7-1 Typical Shaped Charge – Cone End

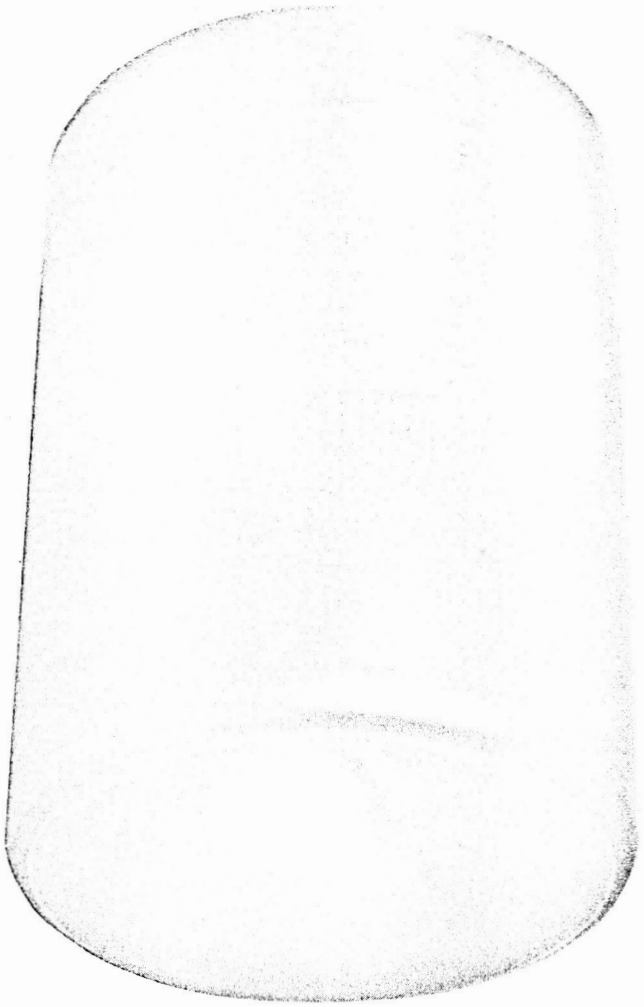


Fig. 7-2 Shaped Charge

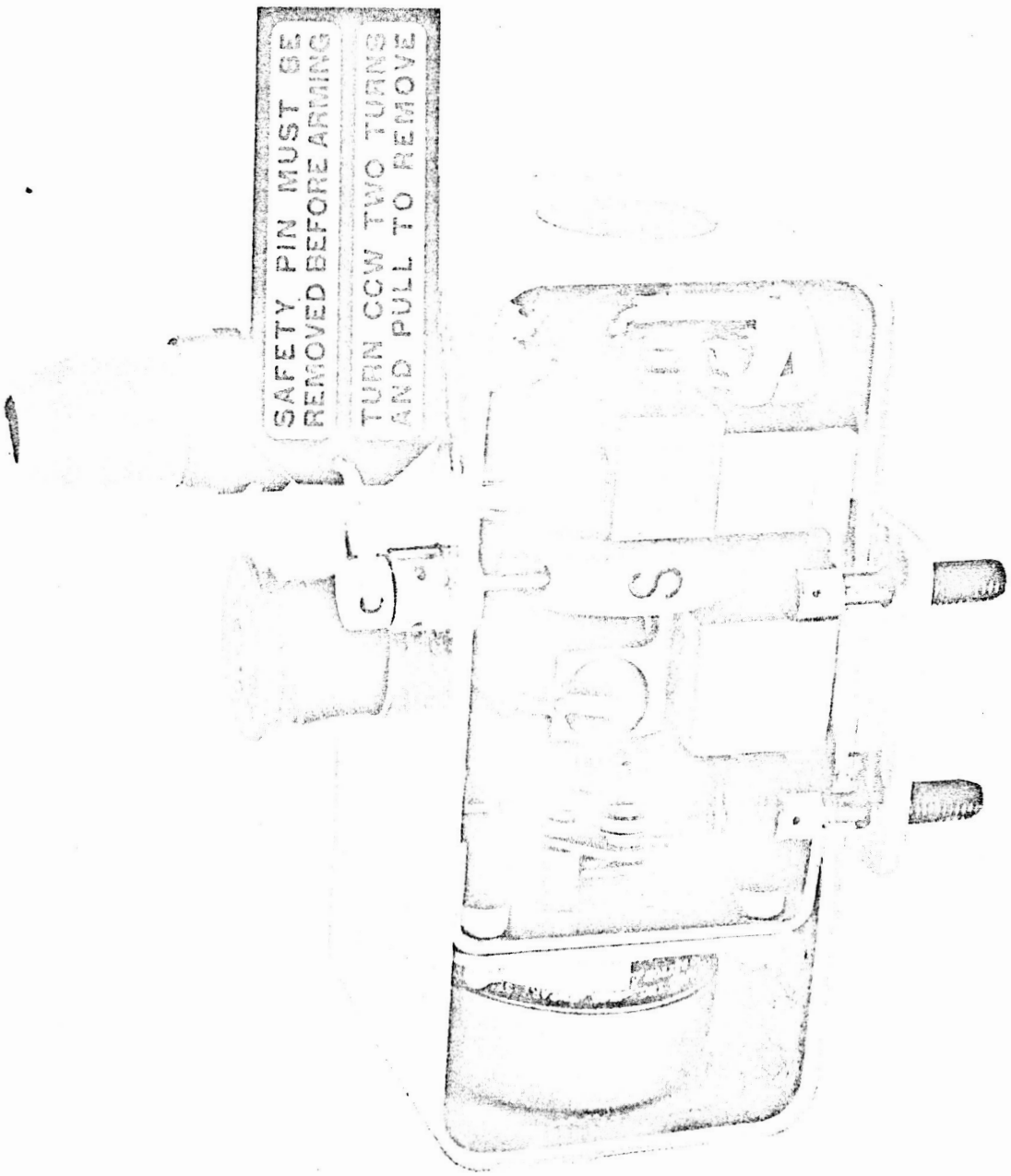


Fig. 7-3 Destruct Initiator

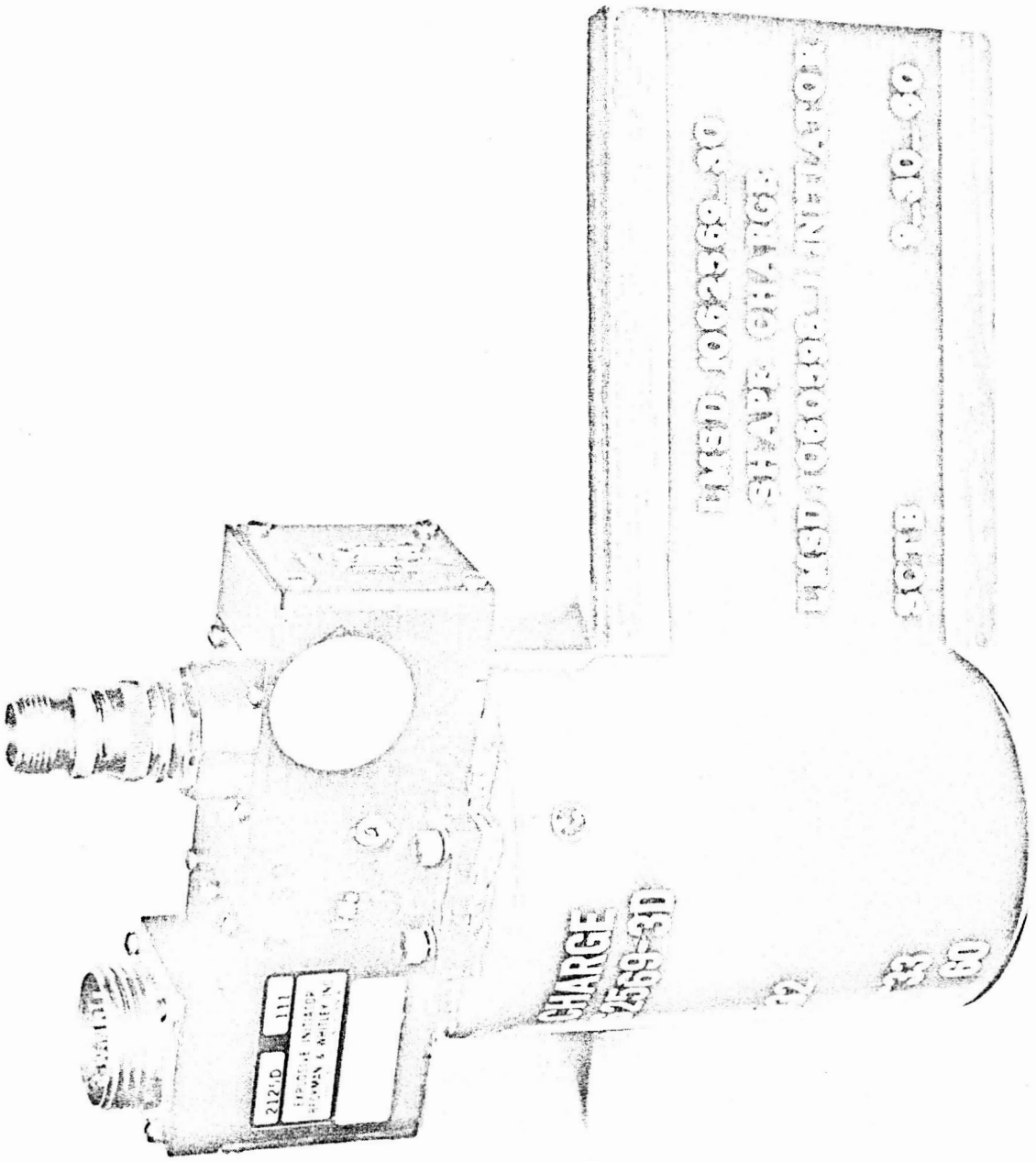


Fig. 7-4 Shaped Charge and Initiator Assembly

A Ledex switch, mechanically linked to the end of the rotor, converts an electrical impulse into a rotary motion. A mechanical latching mechanism (ball detent) assures that the rotor positively positions itself at 90-deg (large initiator) or 180-deg (small initiator) intervals. Repeated activation of the Ledex switch successively rotates the rotor 90 deg (safe-arm-safe-safe) or 180 deg (safe-arm-safe) and rotation is always in the same direction.

To lock the rotor in the safe position, a safety pin is inserted through the explosive initiator assembly and a hole in the rotor. This safety pin is removed after the destructor has been electrically connected and all vehicle checks have completed.

A window in the end of the explosive initiator assembly permits a visual check of the rotor position.

Arming Circuit – The arming circuit is designed for remotely arming or disarming the satellite flight termination system. Remote control is from the blockhouse and is accomplished by advancing the rotor-Ledex assembly of the explosive initiator to the arm safe positions. Arm and disarm functions are switch-operated by wafer switches connected to the rotor-Ledex assembly. The wafer switches function like a two-position latching relay when the rotor assembly is rotated; only the mechanical rotation of the rotor actuates the wafer switches. The arming circuit is also isolated electrically from the battery-destruct circuit by this arrangement. In the safe position, the bridgewires in the explosive detonator are shortened, and the auxiliary battery leads which supply the input signal current are electrically isolated. In the arm position, the shorts across the bridges are opened and the bridges are electrically connected to the input signal leads.

Electrical Characteristics – The explosive detonator is provided with two bridgewires, each capable of initiating an explosion. Characteristics of the detonator are presented in Table 7-4.

Table 7-4

SELF-DESTRUCT SYSTEM DETONATOR
ELECTRICAL CHARACTERISTICS

Bridgewire Resistance	0.8 to 1.2 ohms (70°F)
Maximum No-Fire Current	0.5 amp for 5 min
Always Fire Current	2 amp at less than 2.5

7.2.3.2 Command Destruct System. In addition to the self-destruct system located on the booster adapter, command destruct receivers, which are optional equipment available for Agena-D vehicles, have been installed to provide a means of permanently shutting down the engine. For Gemini vehicles, the explosive charge is removed, leaving only the antennas and receivers. The system is actuated upon receipt of a ground command through a redundant system of two antennas and two Agena-mounted command destruct receivers. The receivers then transmit a pulse through the flight control J-box to the aft safe/arm J-box where two pairs of relays are tripped, the engine start signal is interrupted, and the engine is shutdown as described in Section 4.

Shutting down the engine with the command destruct system permanently disables the Primary Propulsion System, because the relay contacts in the aft safe/arm J-box cannot be reset. Therefore, once the vehicle mission has been completed, the command destruct system can be actuated and the vehicle can not be fired by any spurious electrical signals.

7.2.4 Booster Retrorocket, LMSC-1062410-5

Upon actuation of the separation charge, the Gemini-ATV is released from the booster adapter. The booster retrorockets provide reverse thrust to reduce the booster velocity relative to the Agena velocity thus causing positive separation. The 0.9 KS-500 solid-propellant retrorocket motors, LMSC-1062410-5, are designed and qualified by Rocket Power, Inc. Each rocket motor develops approximately 500 lb of thrust for a nominal burn

time of 0.9 sec. The maximum length of each unit is 15.25 in. and the maximum diameter is 2.9 in. (See Fig. 7-5 for a cutaway view.) The motor case is a cylindrical assembly of AISI 4130 alloy-steel, having a minimum yield strength of 135,000 psi. Nozzle and forward closure assemblies are also of AISI 4130 steel with a graphite insert in the nozzle throat. For thermal control, the case, forward closure, and the outside of the nozzle are either coated with white epoxy paint or nickel plate. Four machined and drilled lugs, welded to each motor case, are used for attaching the rockets to the booster-adapter mounting pads. The rockets are mounted 180 deg apart on the Z-Z axis, with the nozzles canted away from the booster adapter; the centerline of thrust runs through the empty CG of the booster. Mounting provisions allow for alignment of thrust with respect to the booster empty CG and adjustment of motor case to X-X axis parallelism. See Table 7-5 for performance characteristics and Table 7-6 for weights and dimensions.

Table 7-5

PERFORMANCE CHARACTERISTICS OF 0.9 KS-500 RETROROCKETS

Propellant	2.1 lb RPI-PAP-8 polysulfide fuel and ammonium perchlorate oxidizer
Duration	0.925 sec nominal*
Thrust	490 lb nominal*
Chamber Pressure	725 psia nominal*
Total Impulse	455 lb-sec*
Ignition Delay	0.013 sec*
Operating Temperature Range	-40° to +200°F
Storage Temperature Range	-20° to +120°F
Auto-Ignition Temperature of Propellant	360°F
Maximum Guaranteed Ignition Altitude	150,000 ft

*At sea level and +70°F.

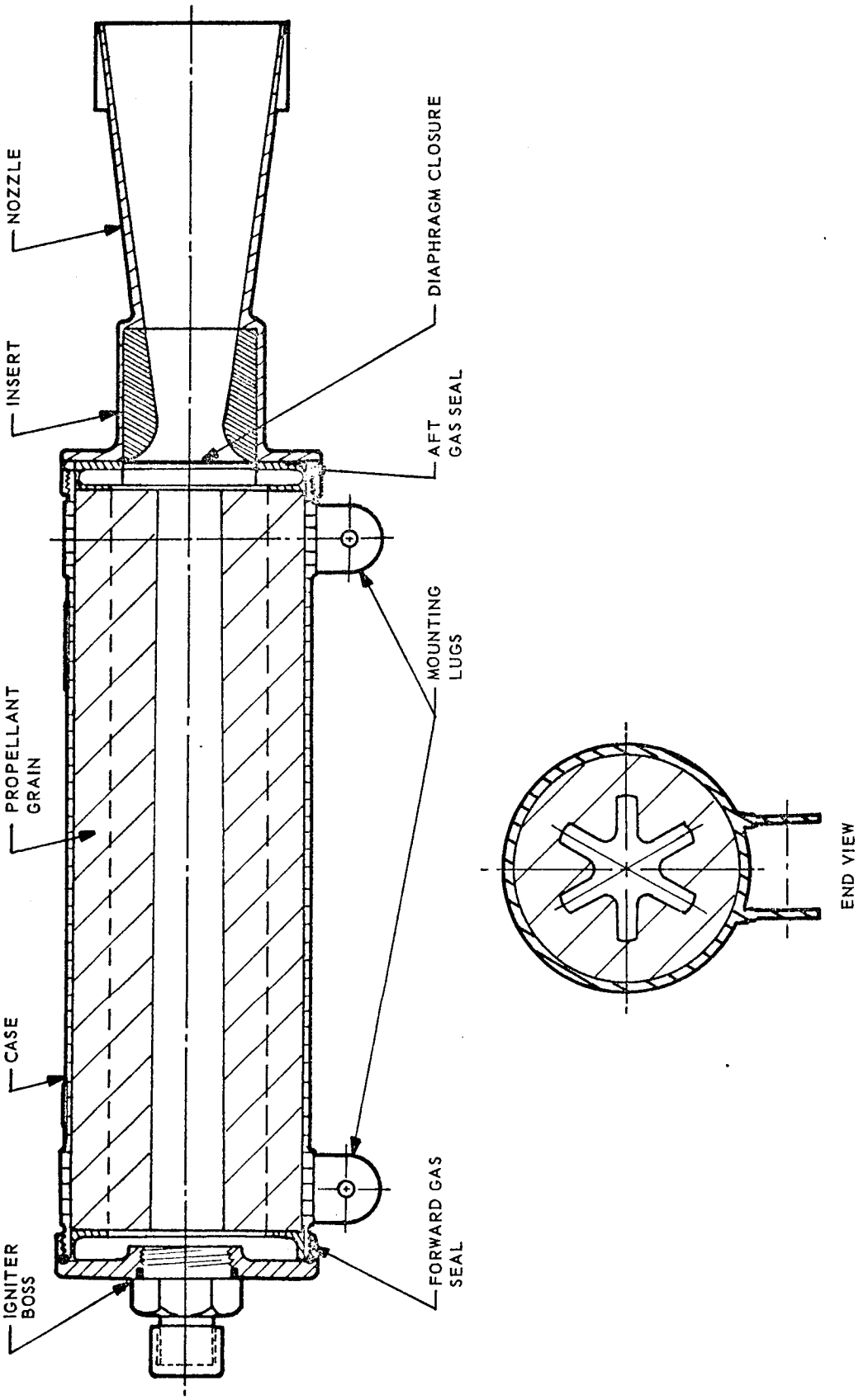


Fig. 7-5 Retrorocket – Cartaway and End View

Table 7-6

NOMINAL WEIGHTS AND DIMENSIONS OF RETROROCKETS

Rocket Motor Dimensions	
Overall Length	15.25 in.
Overall Diameter	2.9 in.
Weight Before Firing	4.69 lb
Weight After Firing	2.56 lb
Nozzle	
Throat Area	0.455 to 0.458 $in.^2$
Expansion Ratio	7.6 to 1
Rocket Motor Weights	
Motor Case	1.20 lb
Propellant Grain and Liner	2.16 lb
Forward Closure	0.36 lb
Nozzle and Insert	0.67 lb
Igniter	0.15 lb
Miscellaneous (gas seals, diaphragm, and inhibitor)	0.15 lb
Total	4.69 lb

Starting from the forward end of the motor, the component parts in order are: the igniter, forward closure, gas seal and O-ring, propellant grain and liner, motor case, aft gas seal and O-ring, nozzle-closure diaphragm, and nozzle assembly with graphite-throat insert.

The igniter is an electrically actuated hot-gas type. The igniter assembly consists of a 1-gm pulverized boron-potassium nitrate booster charge and a 2-gm pressured boron-potassium nitrate sustainer charge, contained in a threaded steel shell which is hermetically sealed to the forward closure. A double bridgewire ignites the booster charge. The igniter mates with a 3106-10SL-4S Bendix-type connector.

The forward closure, including the hermetically sealed threaded-end igniter, is screwed to the forward end of the motor, and an O-ring seal, in combination with sealant applied to the threads, completes the sealing process on the forward end.

A hot-gas seal, bonded to the forward end of the grain, limits the amount of hot gas which is applied to the threads and outer periphery of the forward propellant grain face. Due to the design, the sealing ability is improved as the pressure increases.

The solid propellant grain is a polysulfide fuel-ammonium perchlorate oxidizer composition with an internal burning, six-pointed star configuration. This composition is bonded to the motor case with a polysulfide slurry mixture which also acts as an inhibitor. The case, which is threaded on both ends to receive the closure and the nozzle, is tested of 3500-psi proof pressure. The aft end gas seal is of the same material as the forward seal and is bonded to the grain.

Between the gas seal and graphite-nozzle insert is a nozzle-closure diaphragm made of 0.030-in. thick aluminum, which is designed for burst at 150 psi. To retain the nozzle insert, the nozzle assembly has a machined well. The assembly is threaded to the case and sealed with an O-ring and sealant on the threads.

Electrical Characteristics – The electrical characteristics of the booster retrorocket are presented in Table 7-7.

Table 7-7
BOOSTER RETROROCKET ELECTRICAL CHARACTERISTICS

Maximum Test Current	10 ma per bridge
Resistance:	
Bridge Circuit	0.7 to 1.3 ohms
Bridge-to-Case	10 meg minimum
Maximum No-Fire Current	0.2 amp for 30 min
Minimum Sure-Fire Current	3 ±0.15 amp for 0.1 sec

7.2.5 Pin Pushers

Pin Pushers are utilized for the purpose of releasing and ejecting the horizon sensor fairings at vehicle separation. A typical pin pusher can be seen in Fig. 7-6. The pin-pusher has a two-piece piston and pin assembly in an aluminum case with pressure cartridges for the two ports. The portion of the pin assembly which fastens to the sensor fairing is retained in the pusher body by a shear pin. The piston and the other half of the pin assembly are mounted in the pin pusher actuating cylinder and positioned by a stop on one side of the piston. As the pressure cartridges are fired, the fairing is released which pushes the piston, breaks the shear pins, and drives the fairing away from the vehicle. Two pin pushers are used on each of two fairings, one located at each end.

7.2.6 Pin Puller

Figure 7-7 shows the horizon sensor torque tube actuator pin puller. Actuation of this pin puller permits the horizon sensor torque tube to change the position of the horizon sensors with respect to the vehicle, since the pin acts as a positioning stop for the torque tube. Removal of this positioning pin, by firing two pressure cartridges in each puller, allows the torque tube and sensors to move to the next position by spring action. The pin puller has a one-piece piston and pin assembly inserted into an aluminum body, containing two pressure cartridge parts and a mounting flange. The pin is drawn into the body by the application of gas pressure to the pin side of the piston.

7.2.7 Separation Bolt

The separation bolt is used for separation of the protective shroud from the Agena vehicle. A view of the double-ended bolt with cartridges installed is

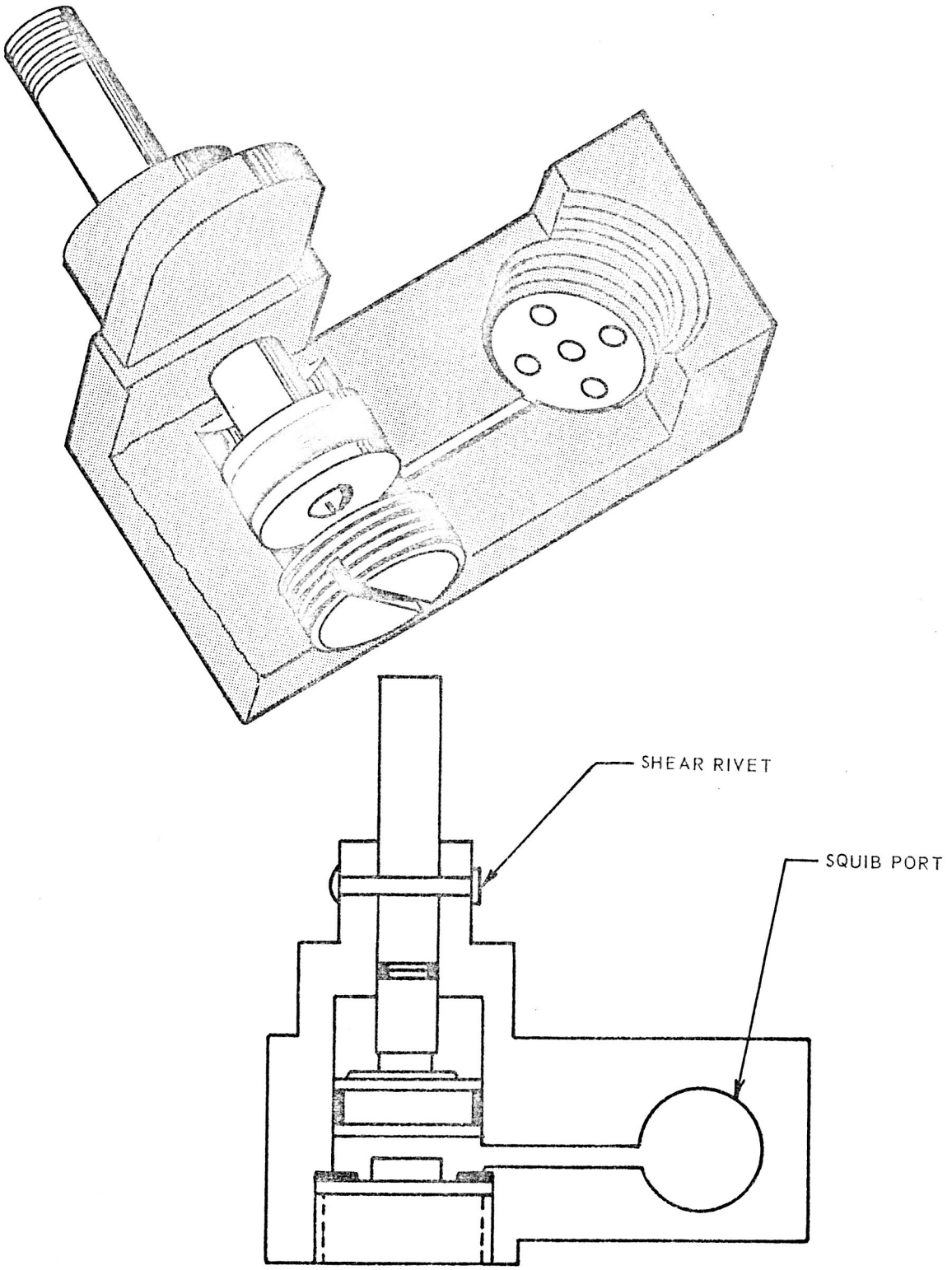


Fig. 7-6 Separation Pin Pusher – Horizon Sensor Fairing-Vehicle

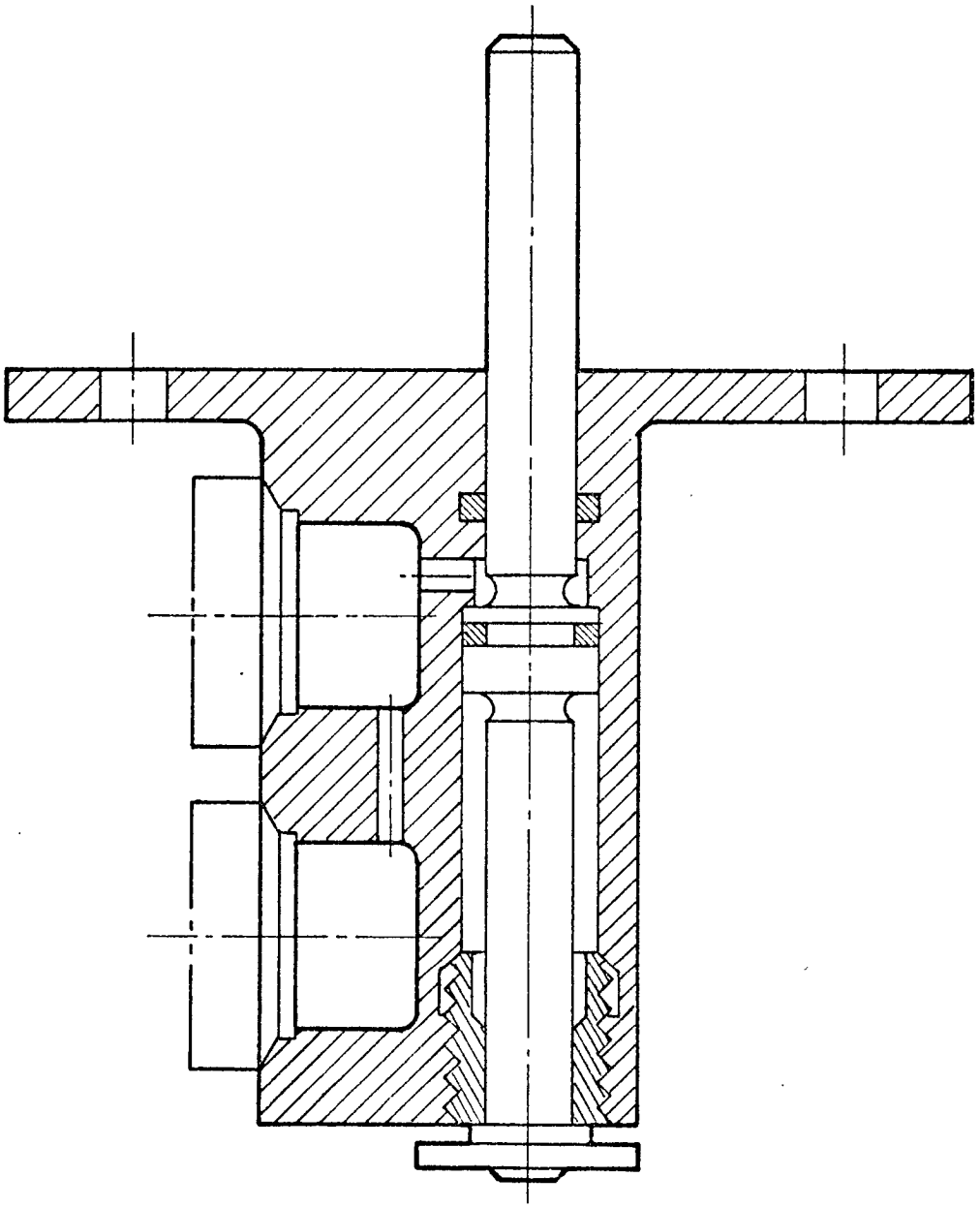


Fig. 7-7 Horizon Sensor Torque Tube Actuating Pin Puller

shown in Fig. 7-8. The bolt is 3.19-in. long, 11/16-in. 24 external threads, and is made from 4340 heat-treated steel. The bolt carries an ultimate load in excess of 16,000 lb. Upon application of firing pulse, the bolt fractures within 10 ms.

The separation bolt is used primarily as a release mechanism and does not provide the energy required for the shroud separation. Fracture of the bolt releases a shroud clamp mechanism and allows the shroud to separate. Since one fracture is sufficient for clamp mechanism separation, the two fracture capability of the double-ended bolts is redundant. Additional redundancy is provided by the dual bridgewire of the detonating cartridge; either bridgewire will cause the cartridge to function.

Detonating Cartridge – The detonating cartridge is intended solely for use with the separation bolt. Upon actuation, bolt fracture is caused by a combination shock-pressure effect. The detonating cartridge is constructed with a 1.310-in. long, 1/2-in. 20-thread output end. Overall length of the cartridge is approximately 1.82-in. The center of the cartridge has a 0.625-in. diameter hexagon protrusion for use with flat wrench. The connector end is manufactured to mate with the Bendix PC06E-8-4S. The base charge is 135 mg of PETN, and the transfer charge is 203 mg of lead ozide. The shell is stainless steel. The electrical characteristics of the detonating cartridge are presented in Table 7-8.

Table 7-8

SEPARATION BOLT DETONATING CARTRIDGE
ELECTRICAL CHARACTERISTICS

Maximum Test Current	10 ma per bridge
Bridge Circuit:	
Pins A-to-D	0.5 ±0.1 ohm
Pins B-to-C	0.5 ±0.1 ohm
Pin-to-Case Resistance	1 meg minimum
Maximum Safe Current	0.5 amp for 10 min.
All-Fire Current	2 amp within 10 ms

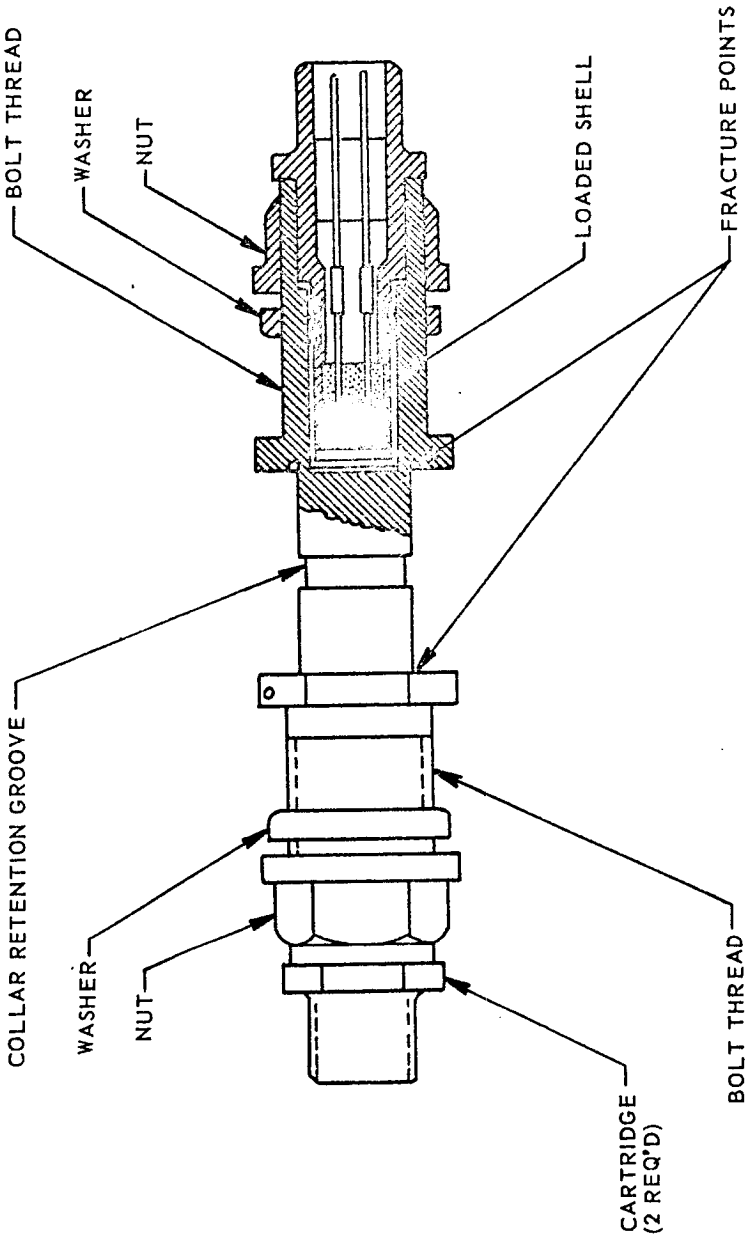


Fig. 7-8 Explosive Shroud Bolt

Section 8
AEROSPACE GROUND EQUIPMENT

8.1 INTRODUCTION

Program-peculiar Aerospace Ground Equipment (AGE) is required to support Gemini Program peculiar requirements at Santa Cruz Test Base, AMR Hangar E, Launch Emplacement 14, and LMSC Sunnyvale. Existing AGE for the Model 8096 engine was inadequate for Primary Propulsion System (PPS) checkout; therefore, new AGE was designed to meet requirements of the Gemini Program. The required AGE is:

- PPS Checkout Console, along with its accompanying tools (Fig. 8-1)
- Start Tank Loading Carts (Figs. 8-2 and 8-3)
- Auxiliary Flushing Unit (start tank flushing) (Figs. 8-4 and 8-5)
- Vacuum Start Tank Bake-out Oven (Fig. 8-6)

8.2 PRIMARY PROPULSION SYSTEM CHECKOUT CONSOLE

The PPS checkout console (Fig. 8-1) is a portable pneumatic-electrical unit utilized in performing the following checkouts:

1. Model 8247 engine functional checks including those for main and gas generator (GG) valve actuations, and for start tanks and dual check valves
2. Model 8247 engine leak checks including those for turbopump seals, main oxidizer system, main fuel system, thrust chamber, turbine exhaust, main valves, and GG valves

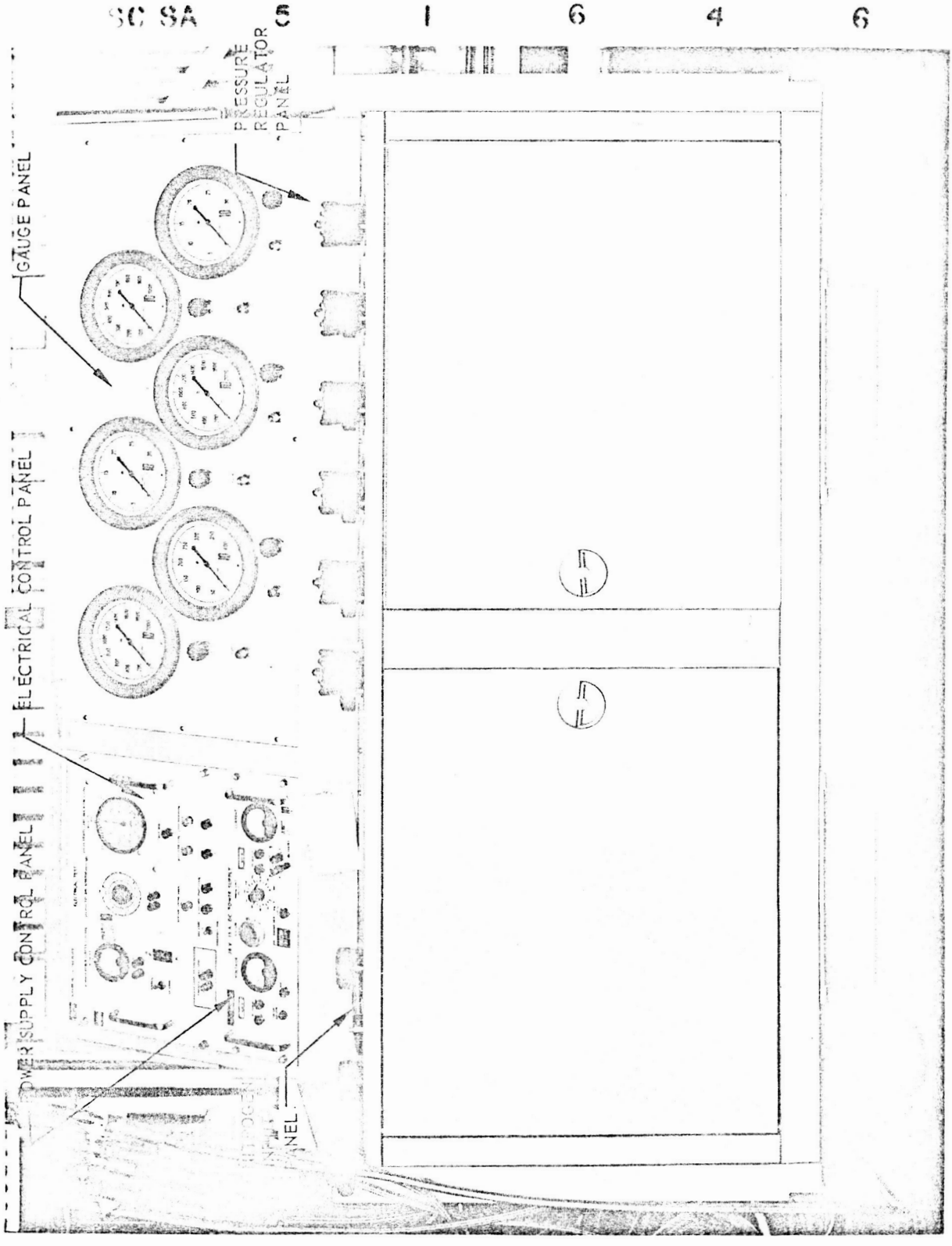


Fig. 8-1 Primary Propulsion System Checkout Console

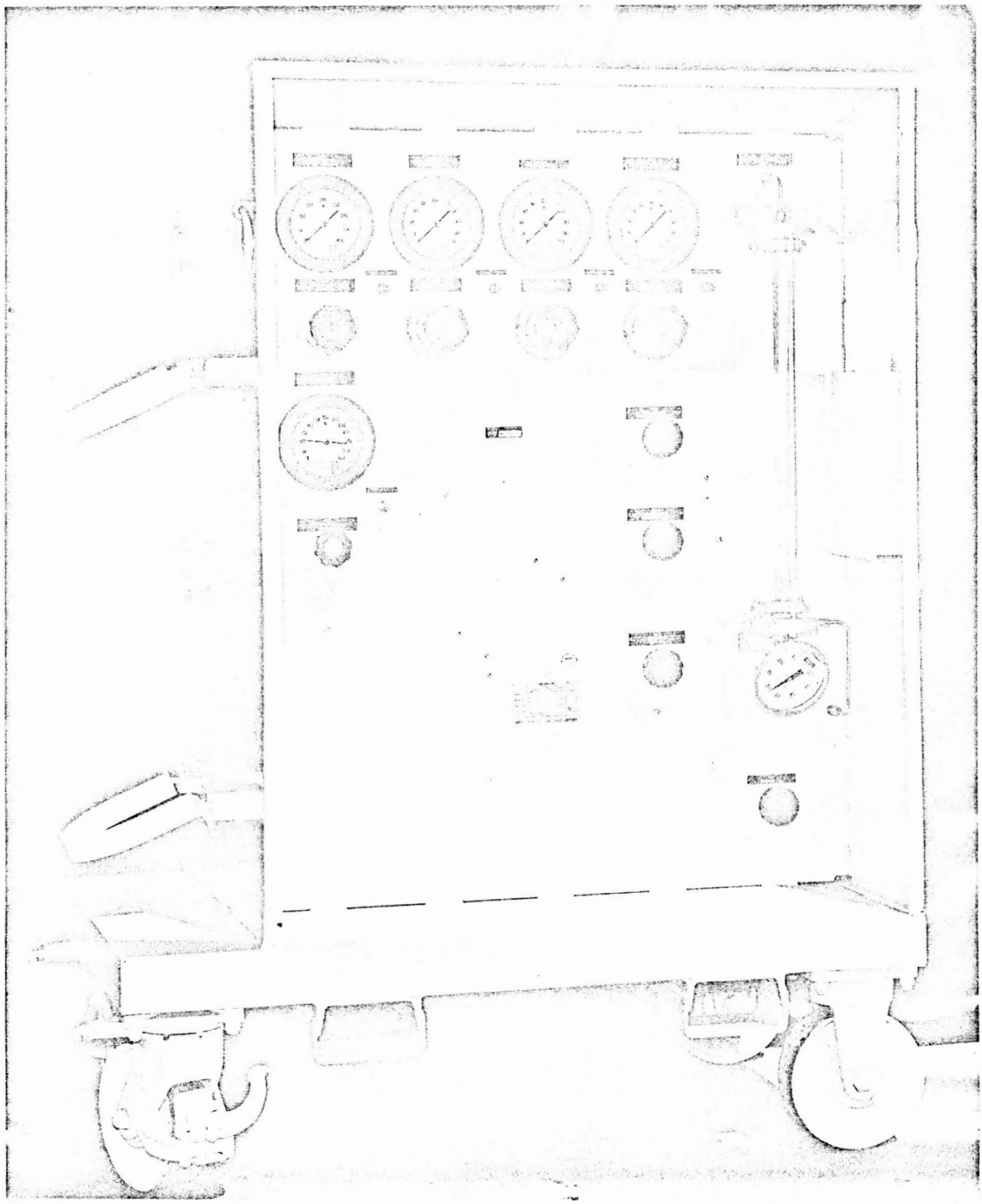


Fig. 8-2 Oxidizer Start Tank Loading Cart

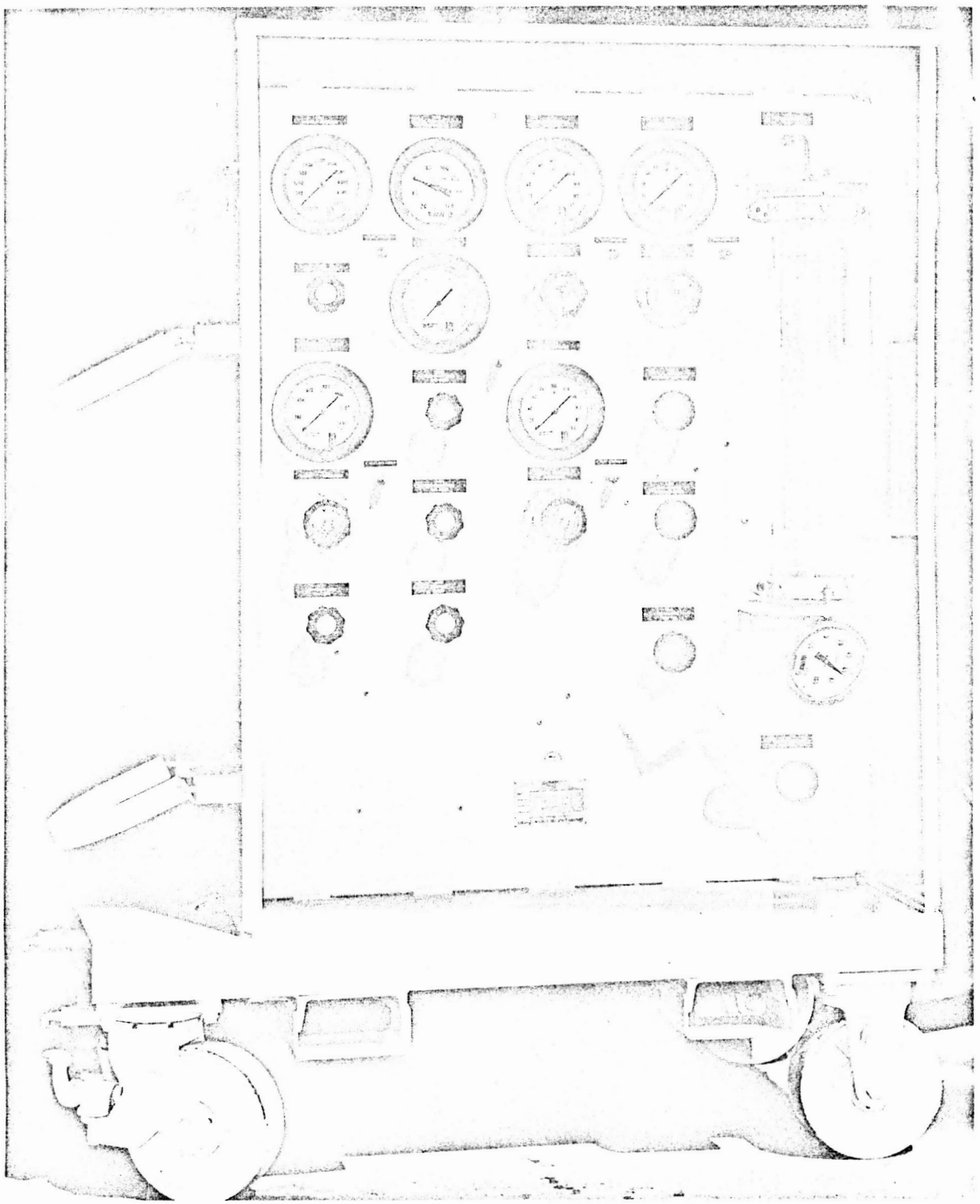


Fig. 8-3 Fuel Start Tank Loading Cart

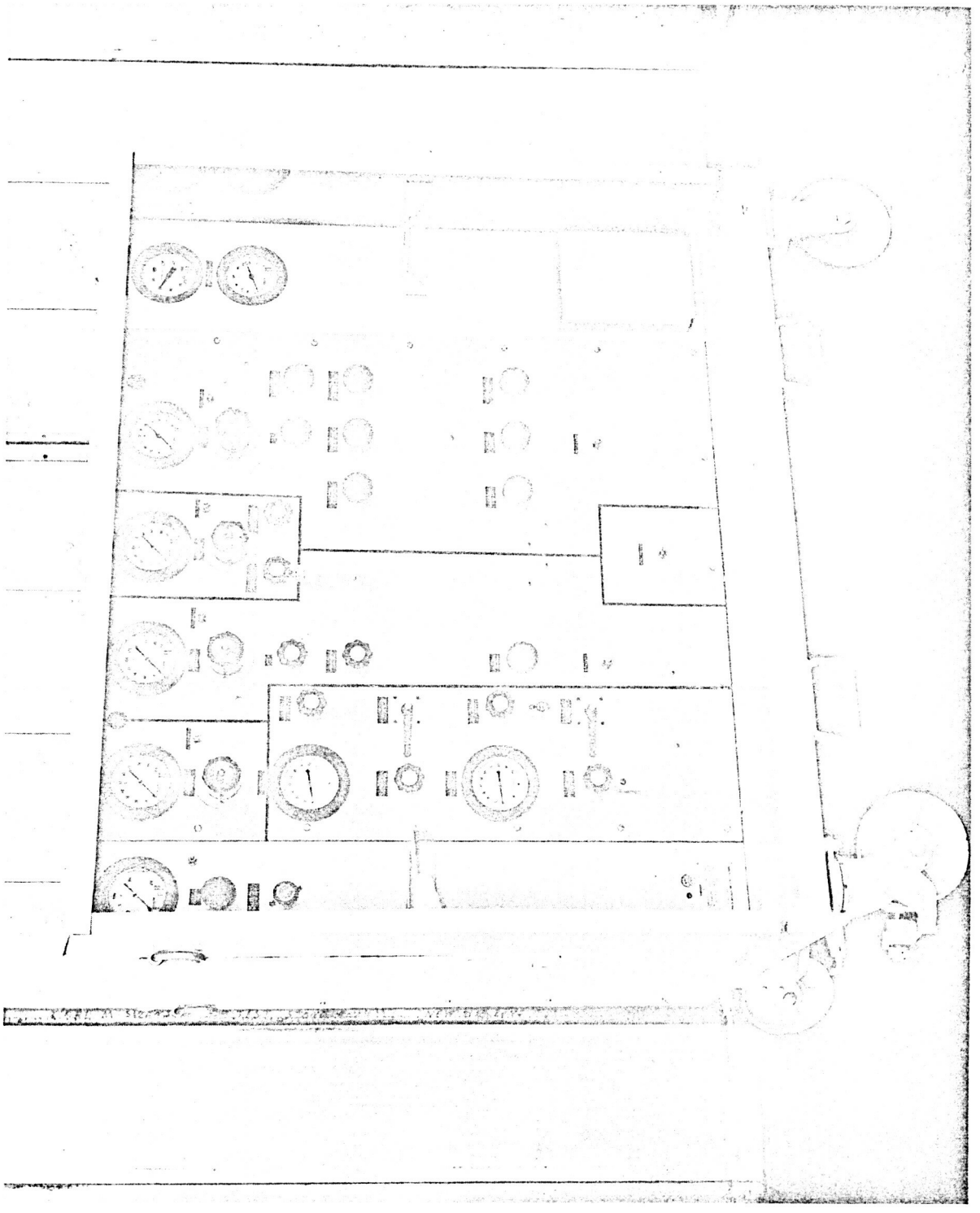


Fig. 8-4 Start Tank Auxiliary Flushing Unit — Front View

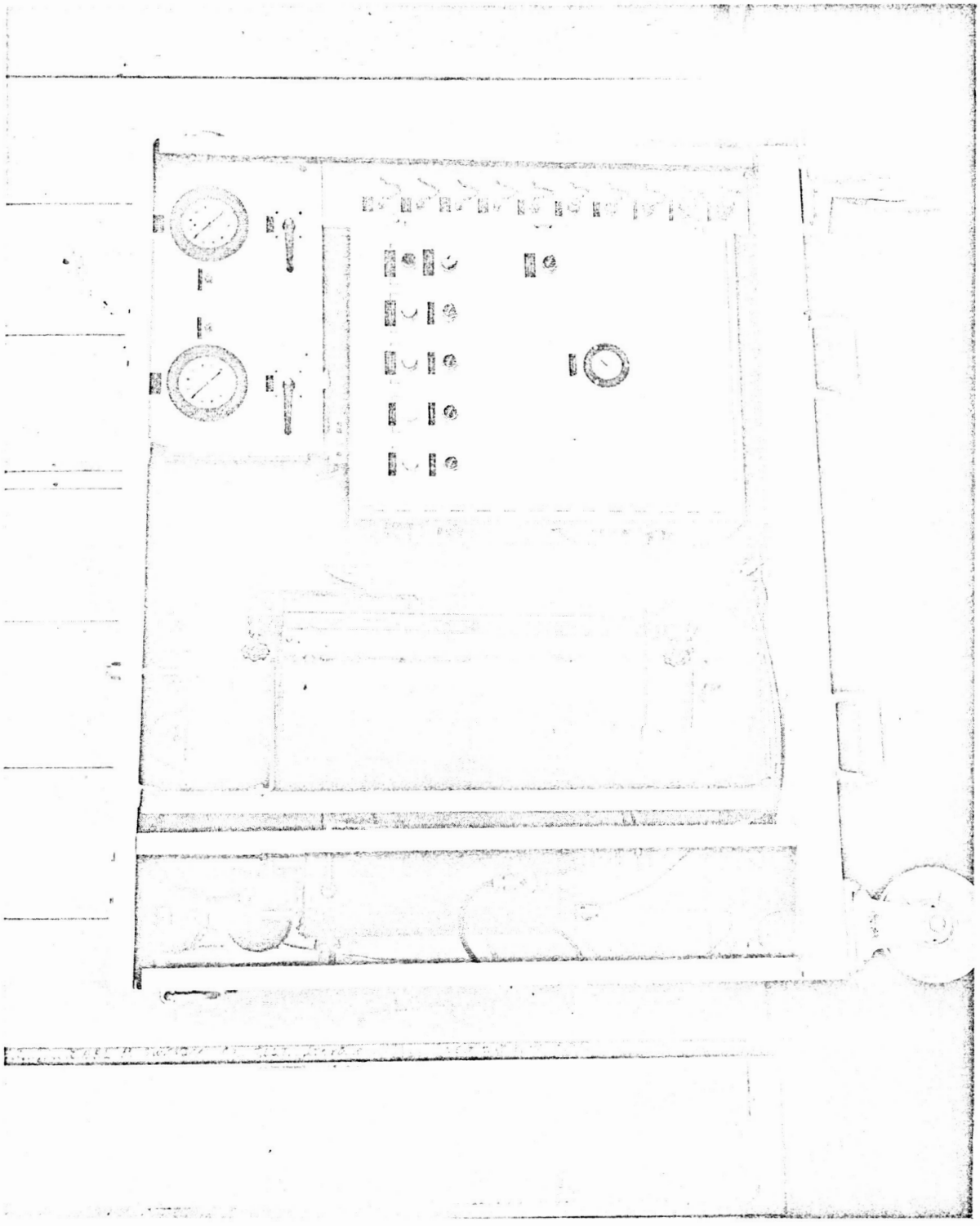


Fig. 8-5 Start Tank Auxiliary Flushing Unit – Back View

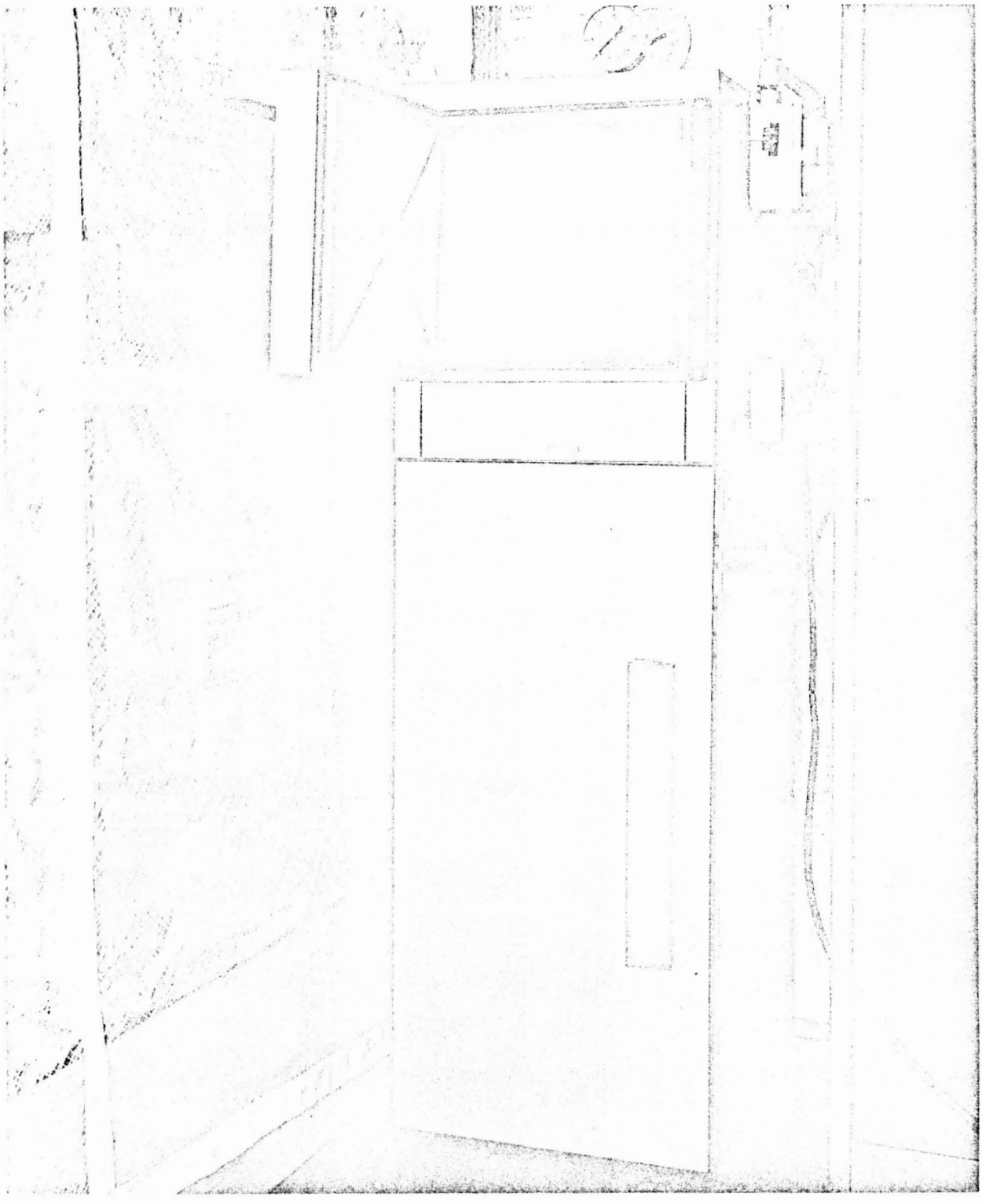


Fig. 8-6 Vacuum Bake-Out Oven (Start Tank)

3. Model 8247 engine turbine overspeed and engine-mounted electronic gate checkouts
4. PPS propellant isolation valve (PIV) functional checkouts

Pneumatic Portion of Cart – Components of each of six separate gas systems (Fig. 8-7) are a relief valve, regulator, shutoff valves, vent valve, isolation check valves, pressure gage, flex hoses, and the necessary interconnecting plumbing. All system outlets protect the engine with integrally mounted-in-line 10-micron absolute filters. Duplicity of systems is required so as not to use the same pressure system for checkout of both fuel and oxidizer components. The pressure ranges of the six systems are 0 to 30 (two systems), 0 to 400, 0 to 800, and 0 to 2000 psig, respectively. Maximum allowed inlet-supply pressure is 3000 psig.

Electrical Portion of Cart – The electrical portion of the cart (Fig. 8-8) consists of circuitry required to perform functional checkout of the electronic gate. Electrical connections are provided to measure the engine "tell-tale" and hence verify proper relay operation within the unit.

The electrical portion of the cart also supplies controls and connections to perform turbine overspeed checkout. Contained within the cart is a means of loading and momentarily shorting the secondary windings (Fig. 8-8) of the two isolation transformers which supply turbine speed signals to telemetry and the Agena status display panel.

Switches, logic circuitry, and a 28-vdc timer are also mounted in the cart and are utilized to time the opening and closing of the PIV's. A design requirement of the timing circuit is that neither equipment malfunction nor operator error shall cause damage to the propellant isolation valves.

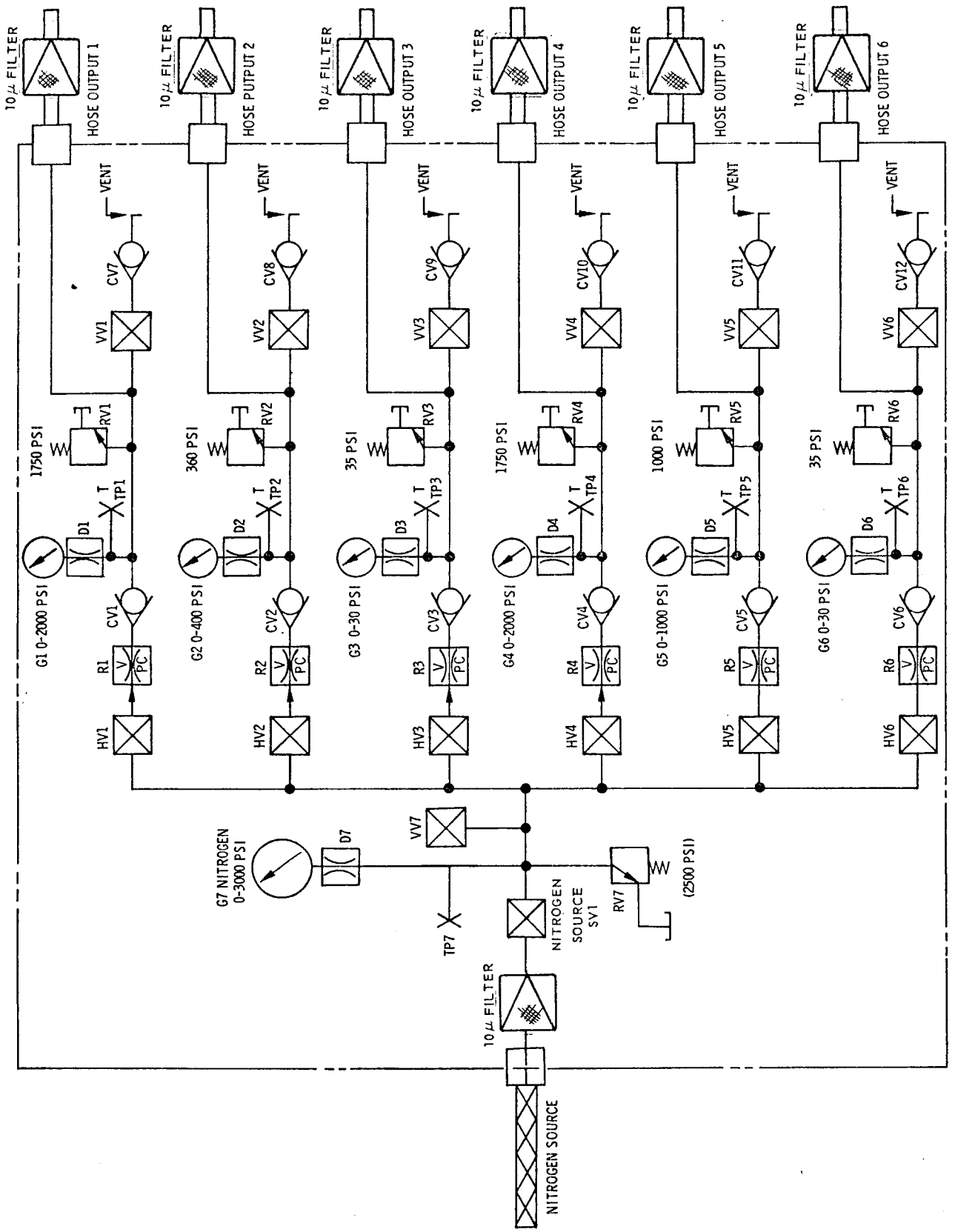


Fig. 8-7 Pneumatic Portion of Primary Propulsion System Checkout Console

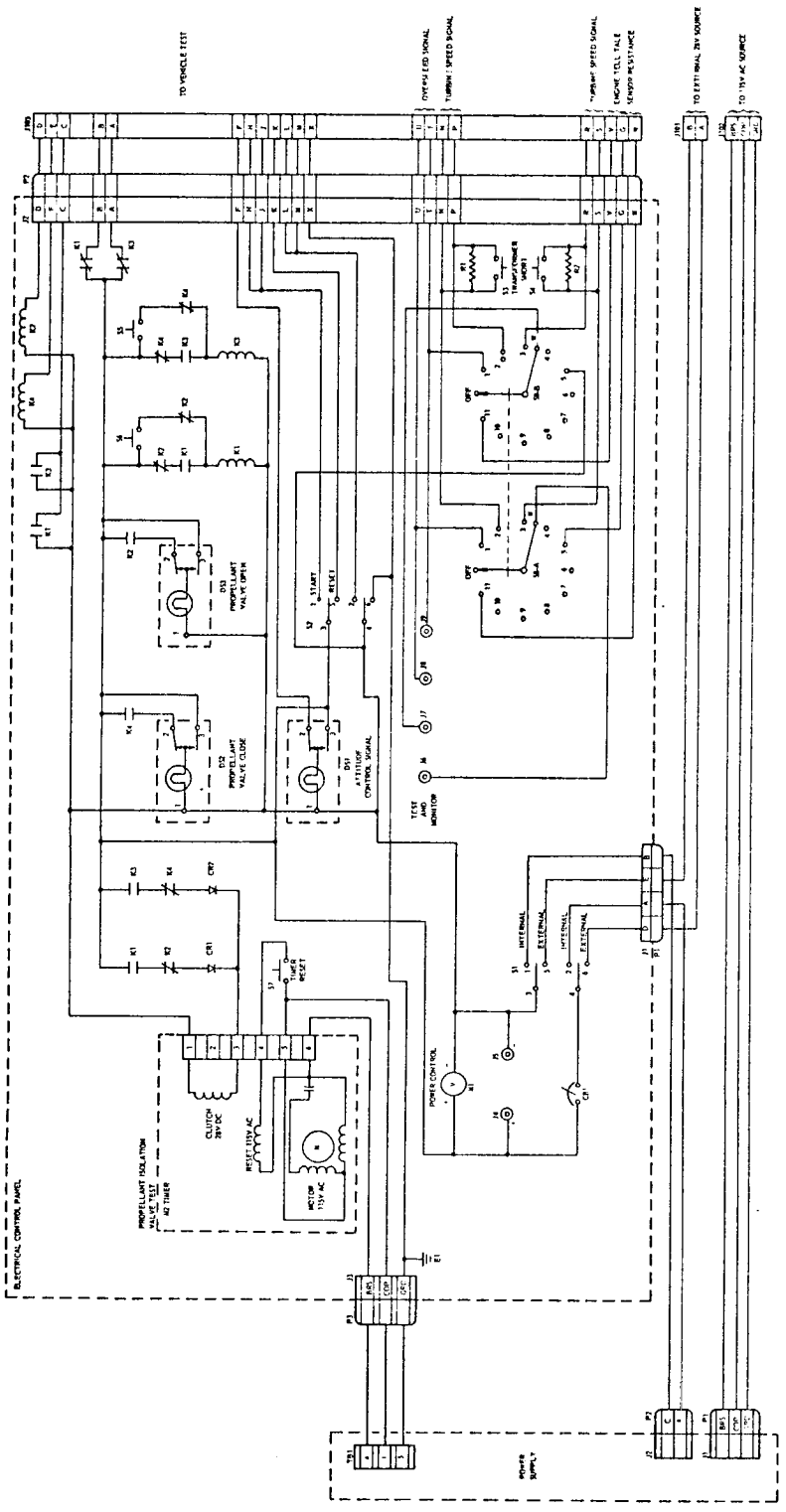


Fig. 8-8 Electrical Portion of Primary Propulsion System Checkout Console

The following tools are supplied in kit form with the PPS checkout console and are used during functional flush and static leak checks of the Model 8247 engine:

1. Thrust Chamber Throat Plug – for plugging the thrust-chamber throat section during thrust-chamber leak check
2. Turbine Exhaust Duct Plug – for plugging the turbine exhaust-duct-exit section during GG and exhaust-duct leak checks
3. Fuel Pump Inlet Cap – for plugging the fuel pump inlet during fuel system pressure and leak checks
4. Oxidizer Pump Inlet Cap – for plugging the oxidizer pump inlet during oxidizer system pressure and leak checks
5. Pilot-Operated Solenoid Valve (POSV) Checkout Tool – for determining poppet leakage of the POSV
6. Oxidizer Valve Actuation Tool – for attaching to the inlet of the oxidizer valve when checking actuation pressure of the valve
7. Oxidizer Venturi Flushing Adapter – for replacing the venturi during oxidizer GG system flush
8. Fuel Venturi Flushing Adapter – for replacing the venturi during fuel GG system flush
9. Oxidizer Valve Flushing Adapter – for providing assistance in overcoming oxidizer valve spring force and providing a port for flush fluid flow
10. Thrust Chamber Throat Protector – for protecting thrust-chamber throat lining during flush.

8.3 START TANK LOADING CARTS

8.3.1 Oxidizer Start Tank Loading Cart

The oxidizer start tank loading cart (Fig. 8-2) is a portable unit (equipped with tires) which contains valves, regulators, relief valves, gages, filters, fuel storage vessel (volumetrically calibrated), and attendant interconnecting plumbing sufficient to accurately load the Model 8247 engine oxidizer start tank. This cart consists of five systems (Fig. 8-9). They are:

1. Pressurization Source System, which consists of the valves and regulators necessary to reduce and regulate source pressure
2. Transfer Tube Pressurization System, which is used to force, by 15-psi regulated gas pressure, a measured volume of propellant into the start tank
3. Fluid Transfer System, which has valving and flex lines utilized to effect the transfer of propellant into and from the AGE-calibrated vessel to the start tank
4. Start Tank Gas Bellows System, which is used to supply "stacking" pressure to the gas side of the start tank bellows immediately prior to start tank load
5. Start Tank Liquid Purge System, which is used to purge the liquid side of the start tank through the start tank fill valve and out the liquid bleed port mounted on the start tank.

All liquid systems on the cart have in-line 25-micron absolute filters, and all gas systems on the cart have 10-micron absolute filters to provide maximum cleanliness to engine start tanks.

8.3.2 Fuel Start Tank Loading Cart

The fuel start tank loading cart (Fig. 8-3) is a portable unit (equipped with tires) which contains valves, regulators, relief valves, gages, filters,

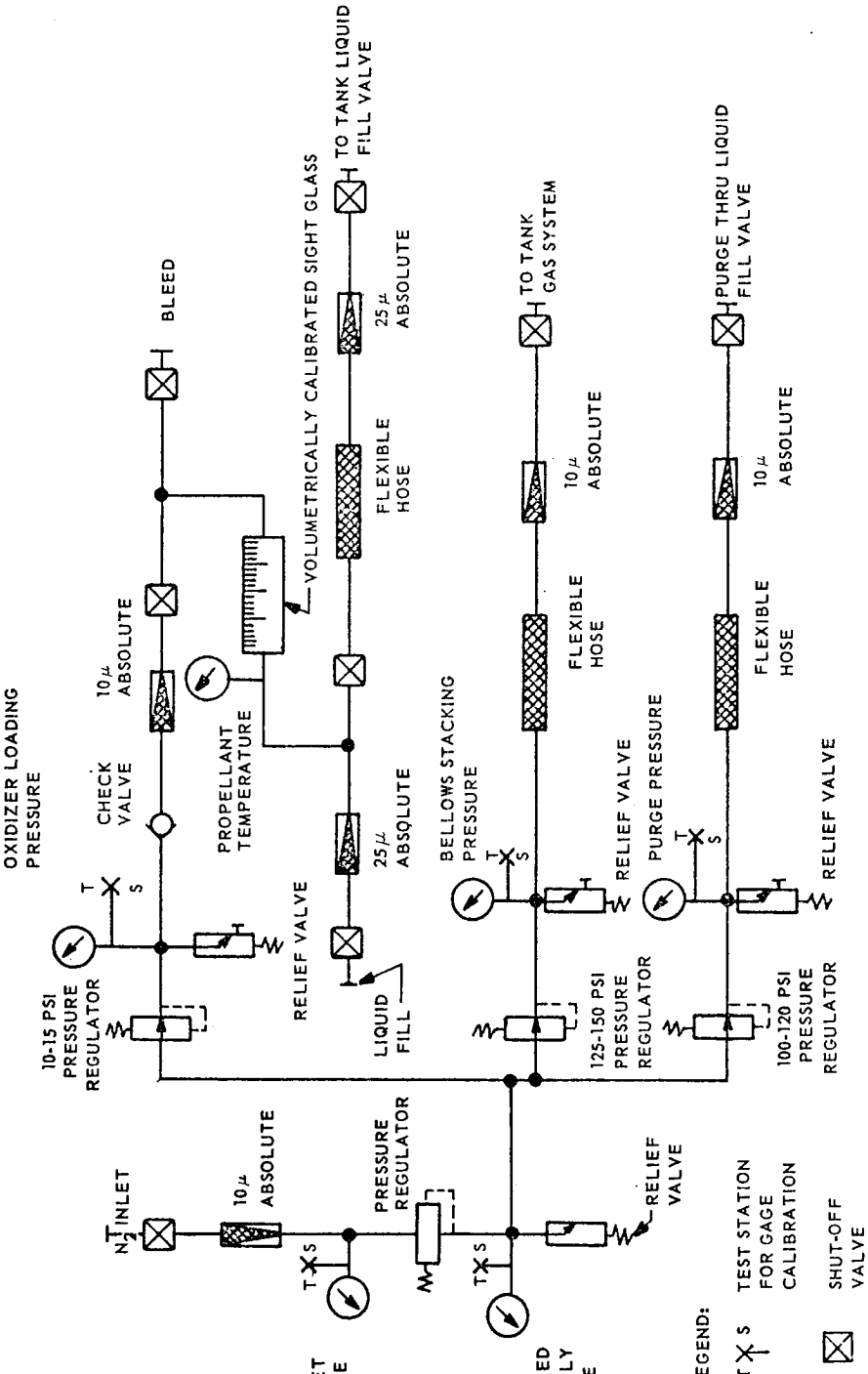


Fig. 8-9 Oxidizer Start Tank Servicing Unit

oxidizer storage vessel (volumetrically calibrated), calibrated gas storage vessels, and attendant interconnecting plumbing sufficient to accurately load the Model 8247 engine fuel start tank and accurately precharge both fuel and oxidizer start tank gas sides. This cart consists of six systems (Fig. 8-10). They are:

1. Pressurization Source System, which consists of the valves and regulators necessary to reduce and regulate source pressure
2. Transfer Tube Pressurization System, which is used to force, by 15-psi regulated gas pressure, a measured volume of propellant into the start tank
3. Fluid Transfer System, which has valving and flex lines utilized to effect the transfer of propellant into and from the AGE-calibrated vessel to the start tank
4. Start Tank Gas Bellows System, which is used to supply "stacking" pressure to the gas side of the start tank bellows immediately prior to start tank load
5. Start Tank Liquid Purge System, which is used to purge the liquid side of the start tank through the start tank fill valve and out the liquid bleed port mounted on the start tank
6. Fuel and Oxidizer Start Tank Precharge, which consists of two accurately calibrated gas volumes and is used to load a predetermined mass of both the fuel and oxidizer start tanks.

All liquid systems on the cart have in-line 25-micron absolute filters, and all gas systems on the cart have in-line 10-micron absolute filters to provide maximum cleanliness to engine start tanks.

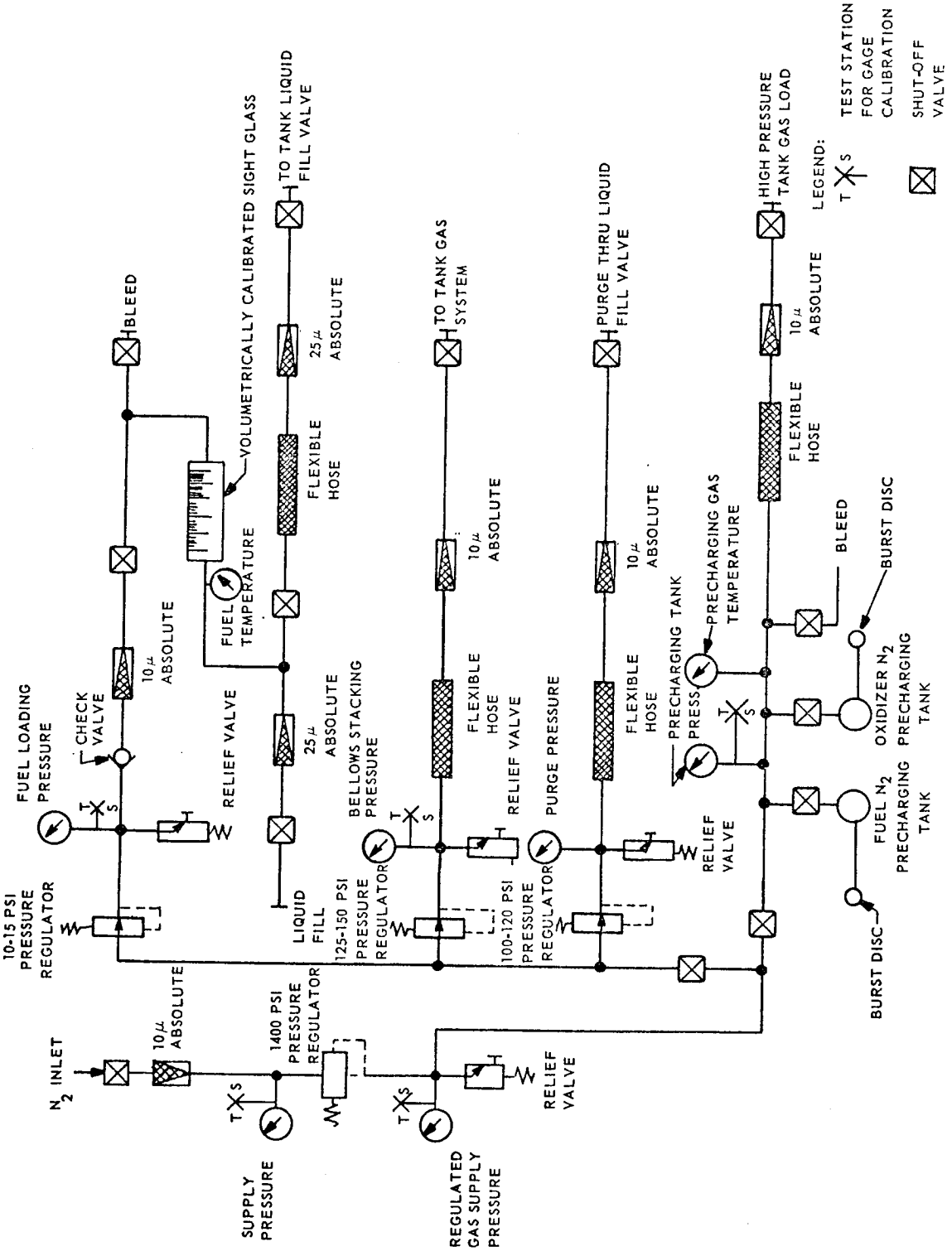


Fig. 8-10 Fuel Start Tank Servicing Unit

8.4 AUXILIARY FLUSHING UNIT (START TANK FLUSHING)

Fuel Start Tank Flushing – The fuel start tank flushing portion of the cart has one storage tank containing methyl alcohol which is pressure fed to the liquid side of the fuel start tank. This tank is cycled in a manner similar to that of the oxidizer start tank – i.e., "pumping" the bellows until all traces of UDMH have disappeared.

Oxidizer Start Tank Flushing – The oxidizer start tank flushing portion of the auxiliary flushing cart (Figs. 8-4 and 8-5) incorporates storage tanks containing heated inhibited water, methyl alcohol, and methylene chloride. Regulators, valves, and interconnecting flex lines are used to pressure feed the tri-flush cleaning agents into the liquid side of the oxidizer start tank. While these fluids are being individually fed into the tank, another AGE system supplies regulated nitrogen to the gas side of the start tank for alternately pressurizing and venting, thereby causing the bellows to move and "pump" the cleaning agent from the tank inlet (liquid-fill valve) and out through the tank-mounted liquid-bleed port. The more highly corrosive IRFNA requires three flushing mediums to gurantee a clean system as opposed to the fuel system which requires one cleaning agent.

8.5 VACUUM START TANK BAKE-OUT OVEN

When flushing of both start tanks has been completed, the tanks are removed from the flushing unit, and with all fittings and ports left open, the tanks are placed in a specially provided vacuum bake-out oven (Fig. 8-6). Then the tanks are baked for four hours at 28 in. of vacuum (approximately 0.7 psia). The baking accomplishes final drying of all start tank internal surfaces.

8.6 MAIN SYSTEM FLUSHING

Main fuel and oxidizer system flushing is accomplished in a manner similar to Model 8096 engine flushing and will not be specifically discussed in this report. Any special hardware required to perform the Model 8247 engine flush accompanies the PPS checkout console and is listed in Paragraph 8.2 of this section.

BIBLIOGRAPHY

1. Agena D Propulsion System for NASA Programs Engineering Analysis Report, LMSC-A372108, 2 April 1963, Confidential
2. Recommended Method for Determining Turbine Back-Pressure, LMSD-419540, 22 December 1958, Confidential
3. Determination of Vacuum Thrust Coefficient, Bell Aerospace Corp., 8048-982-005, 27 October 1958, Confidential
4. Sutton, G. P., Rocket Propulsion Elements, Second Edition, Wiley and Sons, 1956
5. Marks, L. S. Mechanical Engineer's Handbook, Fourth Edition, McGraw Hill, 1941
6. Model Specification Engine Rocket, Liquid Propellant USAF Model XLR-81-BA-13, LMSC-1414463, 22 July 1963, Confidential
7. BAC Model 8247 Service and Maintenance Handbook Liquid Propellant Rocket Engine, Report No. 8247-954201, 31 January 1964
8. Gemini Agena Target Vehicle Familiarization Handbook, LMSC-A602521, 1 April 1964
9. Gemini Agena Target Vehicle Parameters Handbook, LMSC-A374366, 1 May 1963, Confidential
10. Propulsion System Test Requirements for Manufacturing and Final Acceptance Tests S-01B, LMSC-1414805, 31 October 1963
11. NASA Propulsion Subsystem Test Requirements Specification (SS01B), LMSC-1415487, 27 April 1964
12. Telemeter System Instrumentation Schedule, Agena/Gemini, LMSC-1352265, 6 March 1964

13. Satellite Pyrotechnics Course No. SS 154, LMSC-A067093, 9 July 1963

14. Retrorocket (Design Specification), LMSC-1414701A, 23 July 1961

15. Bolt, Explosive Dual Cartridge (Design Specification), LMSC-1412344D, 16 June 1961

16. Specification, Squib - Pressure - Altitude, LMSC-1415503, 1 November 1962

17. Specification - Self Destruct Charge, LMSC-1067259H, 1 November 1961

18. Specification - Umbilical Coupling, Quick Disconnect, Helium and Nitrogen Fill, LMSC-1068731D, 23 June 1960

19. Process Specification (LMSC), Lines and Fittings, Gas and Liquid; Cleaning of, LAC (S) 1481B, 8 September 1963

20. Status Report, Gas Fill Valve, LMSC-A635670, 25 February 1964

21. Requirement Specification, Agena Target Vehicle Propulsion System Test Requirements, LMSC-1416537, 17 April 1964

22. Weight and Performance Status Report - Gemini - Agena D, SP-129-64-2, 1 April 1964

23. Nozzle Extension BAC Model 8247 Design Analysis Report, Report No. 8247-910015, 4 October 1962

24. Design Control Specification, Remote Operated Quick Disconnect Type Propellant Umbilical Connectors, LMSC-1067287, 20 July 1964

25. Specification - Propellant Vent Couplings, LMSC-1510458 E, 26 February 1960

26. Instrumentation Description Agena Target Vehicle, Technical Memorandum 91-35-2, 13 December 1963

27. Propulsion System, Secondary, Liquid Propellant, BAC Model 8250 Engineering Analysis Report, Report No. 8250-910002, 4 January 1964

28. Vehicle Umbilical and Test Plug Pin Assignments, LMSC-A068386, 3 April 1964

29. Agena Target Vehicle Status Display Panel Description, LMSC-A602638

30. Propulsion System Analysis and Preliminary Design Report for Agena/Gemini Mission, Report No. LMSC-A055780, 27 July 1962, Confidential